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13.SPECIAL STUDIES AND STRATEGIC ASSESSMENTS

Since publication of DRA 5.0 several assessments and special studies which are pertinent to defining future human exploration of Mars have been conducted. The descriptions provided in this section represent only a summary of the respective studies. For further information on the studies themselves as well as helpful supporting data can be found in the citations provided in the bibliography section.

13.1. Orbital Missions

Recent discussions within the exploration community have focused on the prospect of the strategy of conducting a mission to orbit Mars as a validation test prior to the surface mission [Augustine, 2009] [1]. Emerging from these discussions is the current National Space Policy that specifically states: *"By the mid-2030s, send humans to orbit Mars and return them safely to Earth."* [Office of the President, 2010] [2] These strategies and conclusions are drawn in part from the historical precedence of Apollo missions where multiple preparatory missions were conducted prior to the first human landing on the Moon. Apollo 8 performed the first human lunar fly-by and Apollo 10 performed the first human orbital mission. Both Apollo 8 and 10 were conducted consistent within the same capabilities and operational profile of the subsequent Apollo 11 landing mission, but that same "orbital testing at the destination before surface landing" philosophy may not hold true for much longer and demanding missions to Mars. Careful examination of the necessary capabilities and knowledge required for both orbital and surface missions, focusing on the similarities between the two, must be conducted to fully understand the potential synergism. To provide a better understanding of how an "orbit only" mission would fit into the emerging strategic framework, an assessment of the operational strategies for exploring the moons of Mars was necessary.

13.1.1. Phobos/Deimos Destination Assessment

Primary Contributor:

Dan Mazanek, NASA, Langley Research Center, USA

13.1.1.1. Introduction

During the first half of 2012, the Human Spaceflight Architecture Team (HAT) developed a preliminary Destination Mission Concept (DMC) to assess how a human orbital missions to one or both of the Martian moons, Phobos and Deimos, might be conducted as a follow-on to human missions to near-Earth asteroids (NEAs) and as a possible preliminary step prior to a human landing on Mars. The HAT Mars-Phobos-Deimos (MPD) mission would also permit the teleoperation of robotic systems by the crew while in the Mars system, hence the hyphenated acronym to emphasize that all three planetary bodies would be explored. The DMC development activity provided an initial effort to identify the science and exploration objectives and investigate the capabilities and operations concepts required for a human orbital mission to the Mars system. In addition, the MPD Team identified potential synergistic opportunities via prior exploration of other destinations currently under consideration.

13.1.1.2. Activity Goal

The primary goal of the activity was to determine whether an opposition-class mission (short-stay mission of ~30-90 days at Mars) provides sufficient time to meet all or most of the science and exploration objectives at Phobos and Deimos, or if a conjunction-class mission (long-stay mission of ~450-540 days at Mars) is required.

Opposition-class (short-stay) missions allow total mission durations that can be significantly shorter than conjunction-class missions (~560 days vs. ~950 days). Conjunction-class (long-stay) missions are "minimum energy" trajectories that require less mission ΔV than opposition-class missions (e.g., ~6.5-7.9 km/s vs. ~8.3-14.1 km/s for crew transfer to and from the Martian system). It should be noted, that the above ΔVs do not include any orbital maneuvers for exploring the Martian system.

13.1.1.3. Background Information on the Martian Moons

Although the origin of the Martian moons has not been conclusively determined, scientists speculate that one or both of the moons are captured asteroids. Regardless of their origin, both moons are relatively small and are similar in appearance to near-Earth asteroids, so human NEA missions could provide applicable operational training that would enable a more efficient future exploration of one or both moons.

Figure 13-1 shows a color image composite of the two moons to the same scale using data obtained by the High Resolution Imaging Science Experiment (HiRISE) camera on NASA's Mars Reconnaissance Orbiter. The color-enhanced view of Deimos, the smaller of the two moons of Mars, was taken on Feb. 21, 2009 and the image of Phobos was taken on March 23, 2008. Excluding the most recent impact craters, Deimos has a smooth surface due to a blanket of fragmental rock or regolith that covers its surface, whereas the surface of Phobos appears to be pock marked with craters, grooves, and linear features. Data from the camera's blue-green, red, and near-infrared channels were combined to generate these color images. Table 13-1 provides a summary comparison of some of the key characteristics of the two moons.

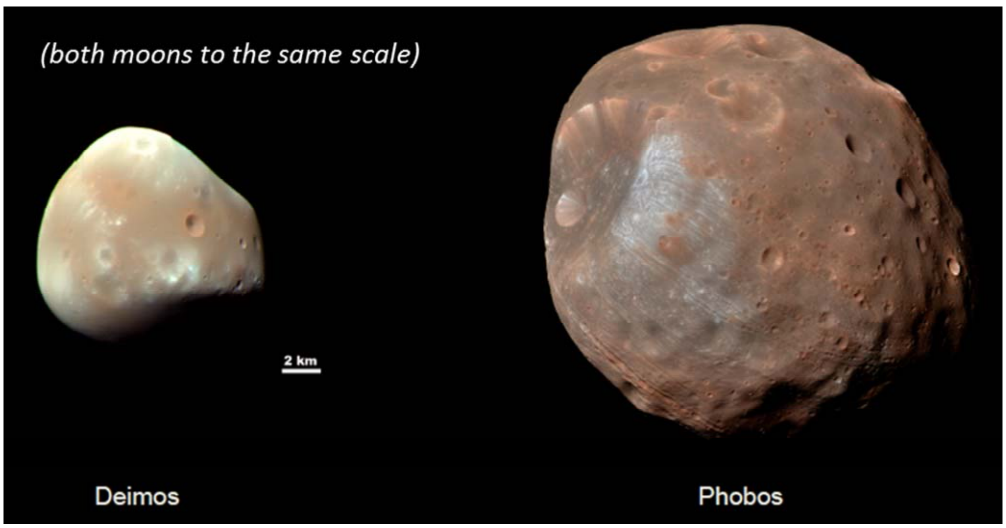


Figure 13-1 Composite Image of Deimos and Phobos (Credits: NASA/JPL-Caltech/University of Arizona).

Table 13-1 Characteristics Deimos and Phobos.

Characteristic	Deimos	Phobos
Mass (kg)	1.80×10^{15}	1.08×10^{16}
Dimensions (km)	15.6 × 12.0 × 10.2	26.2 × 22.2 × 18.6
Albedo	0.068	0.071
Equatorial Surface Gravity (µg)	400	860–190
Semi-Major Axis (km)	23,459 (Mean)	9,378 (Mean)
Inclination to Mars Equator (deg.)	0.93	1.09
Rotation Period (days)	1.26 (Synchronous)	0.32 (Synchronous)

13.1.1.4. Notional Destination Mission Concepts

Figure 13-2 and Figure 13-3 provide notional destination mission concepts for short-stay and long-stay Mars vicinity operations, respectively. These operational concepts provide a graphical representation of the orbital sequencing, examples of possible surface activities, and summary of the Mars orbit strategy. These strategies are examples of how operations could be conducted and are just two of several options that could be developed. The focus of the MPD DMC study effort was to develop a "proof of concept" for the short-stay mission, rather than a definitive baseline. Further refinement of the mission objectives and optimization of the Mars vicinity operations is needed before a final Design Reference Mission (DRM) can be adopted.

While the mission concepts have some similarities (e.g. both mission concepts capture into a 1-sol parking orbit), there are several key differences. In order to minimize mission risk, the short-stay mission performs a plane change maneuver to match the departure asymptote. This is done to assure that the departure orbital conditions are properly set before exploration activities commence. The short-stay mission begins with an exploration of Phobos, which is identified by the MPD DMC Science Objectives and Requirements Formulation team as the higher priority, based on the current state of knowledge of both moons' physical characteristics. The Mars Transfer Vehicle (MTV) is left in the parking orbit with two crew members, while the other two crew members utilize a Space Exploration Vehicle (SEV) attached to a transfer stage to explore the Martian moon and return to the MTV. If sufficient stay time in the Martian system is achievable, the two crew members who remained in the MTV utilize a second SEV and transfer stage to explore Deimos and return. The long-stay mission begins by exploring Deimos with the entire vehicle stack (MTV and two SEVs) transferring to the moon. After orbital operations at Deimos are complete, the entire stack transfers to Phobos. The propulsive element(s) for the long-stay mission must be capable of propelling the entire vehicle stack through all of the maneuvers prior to departure for Earth. In both mission concepts the SEVs are jettisoned prior to the departure maneuver; however, the potential exists for the SEVs to continue operations in an uncrewed mode after Earth departure, limited by their remaining propulsive capability. For the long-stay mission, both SEVs would be in the vicinity of Phobos, whereas for the short-stay mission they would be left in the 1-sol parking orbit.

13.1.1.5. Study Areas

The HAT MPD activity focused on the following seven study areas: 1) science objectives and requirements formulation; 2) exploration objectives and requirements formulation; 3) destination activity implementation strategy; 4) mission implementation strategy; 5) synergies with cis-lunar activities; 6) synergies with human and robotic precursor missions to NEAs; and 7) robotic precursor requirements for a human mission to Mars orbit and its moons. The primary MPD DMC study effort team members and their affiliations are listed at the end of this addendum section and the team members for each study area are in this section.

13.1.1.5.1. Science Objectives and Requirements Formulation

Team: David Beaty (Lead), Paul Abell, Deborah Bass, Julie Castillo-Rogez, Tony Colaprete and Ruthan Lewis

Charter: Identify and prioritize the scientific objectives and requirements for a Mars orbital mission, including small body origin/geology and field science through sampling and geophysical station deployment, Mars geology through the possible collection of Martian meteorites from Phobos, and completing the Mars Sample Return (MSR) mission by retrieving the sample cache from low Mars orbit after ascent from the surface. Additionally, identify possible science opportunities during the transit to and from the Martian system.

Key Findings: The study of Phobos and Deimos contributes Mars Exploration Program Analysis Group (MEPAG) objectives, all science themes in the Small Bodies Assessment Group (SBAG) Roadmap, and includes opportunities for other science activities during transit to and from the Mars system (e.g., astrophysics, heliophysics, life science, etc.).

The highest-priority science is based on sample return and deployment of assets, taking advantage of human crew:

- Small body origin/geology: field science, sampling, geophysical station deployment.
- Mars geology: search for Martian meteorites on Phobos.
- Collect MSR sample cache.

The primary science objectives identified by the study area team were:

- Determine the nature of the surface geology and mineralogy of Phobos/Deimos.

- Characterize the regolith on Phobos/Deimos, and interpret the processes that have formed and modified it
- Complete the MSR Campaign by capturing and returning to Earth the orbiting cache of samples
- Collect any identified Mars meteorites/material from the surface of Phobos/Deimos, and return to Earth for detailed study
- Determine the absolute material ages and constrain the conditions of formation of Phobos and Deimos

In order to determine an exploration scenario that could accomplish these science objectives, notional landing and sampling sites were identified (see Figure 13-4) and used to formulate the destination operational timelines developed by the Destination Activity Implementation Strategy study area team. Table 13-2 identifies the key implementation implication for each of the science priorities.

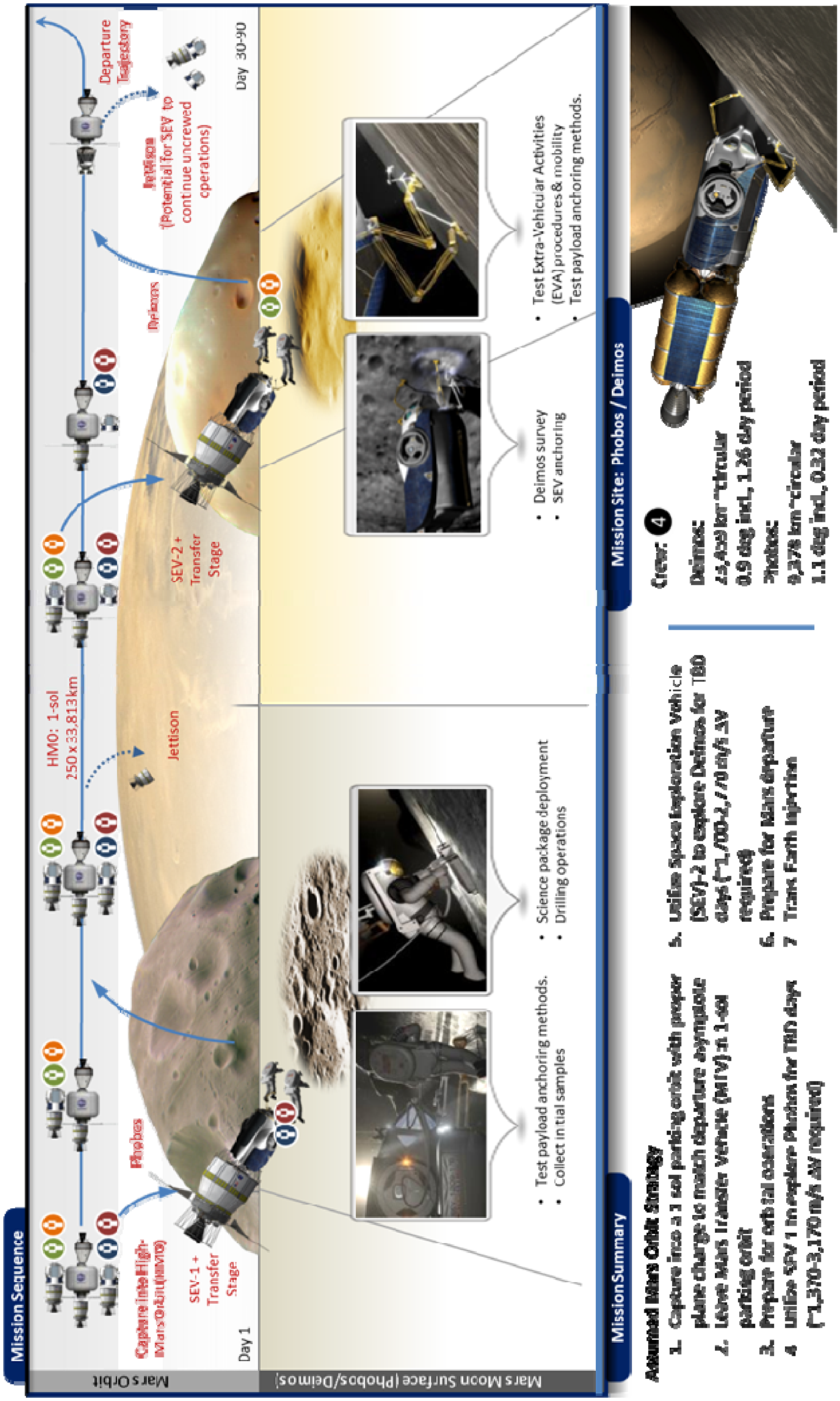


Figure 13-2 Notional concept for short-stay Mars vicinity operations.

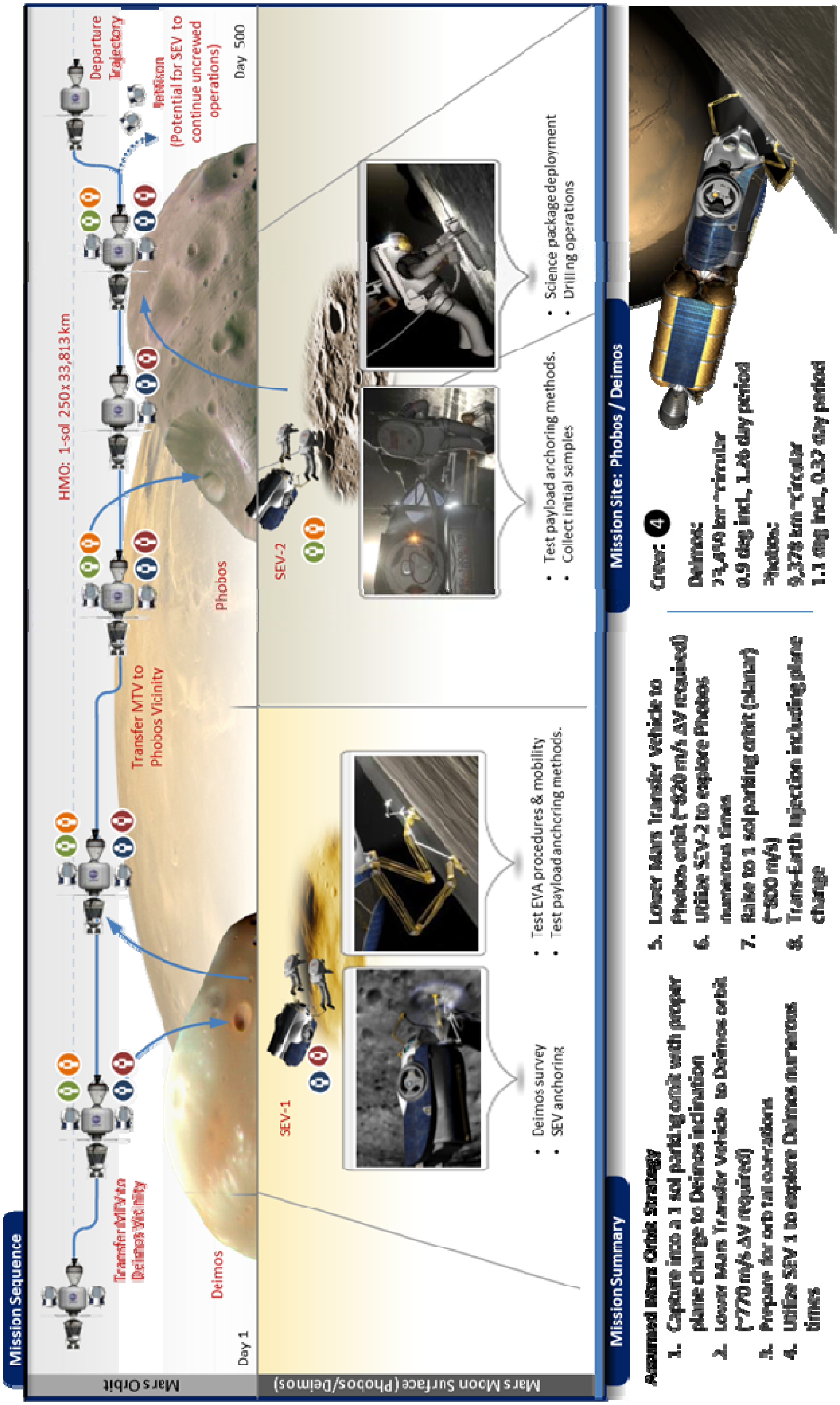


Figure 13-3 Notional concept for long-stay Mars vicinity operations.

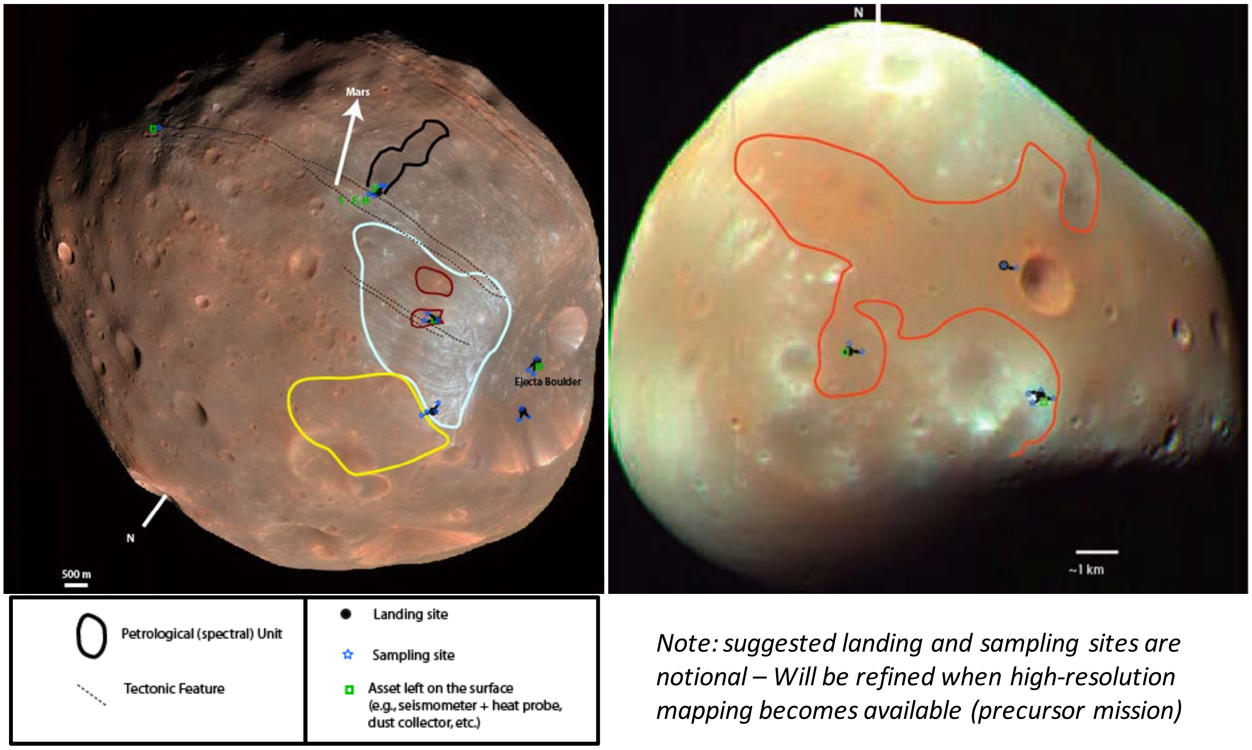


Figure 13-4 Notional reference landing and sampling sites on Phobos and Deimos.

Table 13-2 Implementation implications of science objectives.

SCIENCE PRIORITY	IMPLEMENTATION IMPLICATION
Field science at surface of Phobos/Deimos	This is more efficiently done via EVAs. There is a need for surface mobility to maximize contact time between the astronauts and the geology to be studied. Multiple sites are needed to sample the surface diversity of the moon(s).
Regolith science	This activity requires a method of examining and/or acquiring samples from depth (~2-3 m). Multiple sites are needed to sample the sub-surface diversity of the moon(s).
Returned sample science	The crew would need field instruments to support sample selection/identification, sample acquisition and sample packaging/containment. Samples from multiple sites are needed to capture the material diversity of the moon(s). The returned mass allocation needs to be accounted for within the mission design (including containers, environmental control, etc.).
Long-term monitoring of the Martian system	A suite of monitoring instruments would need to be set up by the astronauts, and left behind on Phobos/Deimos for extended operations.
MSR Sample Cache Retrieval	It is unclear whether this should be achieved via autonomous robotic rendezvous, operation of teleoperated assets, or performed directly by the crew. This requires further study and analysis.
Telerobotics to the Martian surface	The priority is unclear—the science drivers are not well defined. Also it is unclear what the implications are for the duration and location of the astronauts required to teleoperate assets on the Martian surface, and the necessity for pre-deployment of those assets. This requires further study and analysis.

In addition to the science performed in the Martian system, many transit science opportunities are possible during a mission to the Mars system and include the following:

- Venus flyby: likely included during an opposition-class mission.
- Other Planetary Science: micrometeoroid monitoring, dust collection, small body flybys, etc.
- Heliophysics: Sun's polar magnetic field and solar wind characterization, deployment and retrieval of GAS can/SPARTAN-like payloads.
- Astrophysics: Observation of Earth as an exoplanet, planetary microlensing events, etc.
- Biomedical: Monitoring the impact of radiation, microgravity, of solar protons and Galactic Cosmic Rays (GCRs) on cellular material, of the human immune system, of muscular and cardiovascular performance.
- Psychological: Monitoring the impact of confinement, stress hormone levels, sleep patterns, response to altered lighting/environments.

Humans can significantly increase the science returned during transit and make real time adjustments during encounter to maximize science without round trip communication delays and sequencing issues. Additionally, crew members are able to deploy/retrieve equipment repeatedly and return it for detailed examination on Earth. Finally, transit science events will enhance the mission's scientific and engineering return, and provide further opportunities for public engagement.

Recommendations: The study team's recommendation is to visit both Phobos and Deimos, with higher priority for Phobos, based on the current state of knowledge of both moons' physical characteristics. Additionally, the study team recommends that a precursor mission includes science observations necessary to inform human exploration planning (e.g., relative science significance of Deimos vs. Phobos) and retire Strategic Knowledge Gaps (SKGs). The applications of teleoperations to Mars surface are unclear and require further study before providing a recommendation.

13.1.1.5.2. Exploration Objectives and Requirements Formulation

Team: Steve Hoffman (Lead), David Beaty, Tony Colaprete, Bret Drake, Ruthan Lewis and Dan Mazanek

Charter: Gather and articulate the exploration goals and objectives for the human exploration of Phobos and Deimos pertinent to Mars exploration, including orbital missions, surface missions, and preparation for sustained human presence in the Mars system. The human MPD mission is assumed to be part of a larger campaign of human exploration of Mars, including its surface, its moons, and the surrounding environment. The goals and objectives of this mission encompass gathering data and demonstrating technologies/operations needed in advance of humans attaining the next level of Mars exploration (See Figure 13-5).

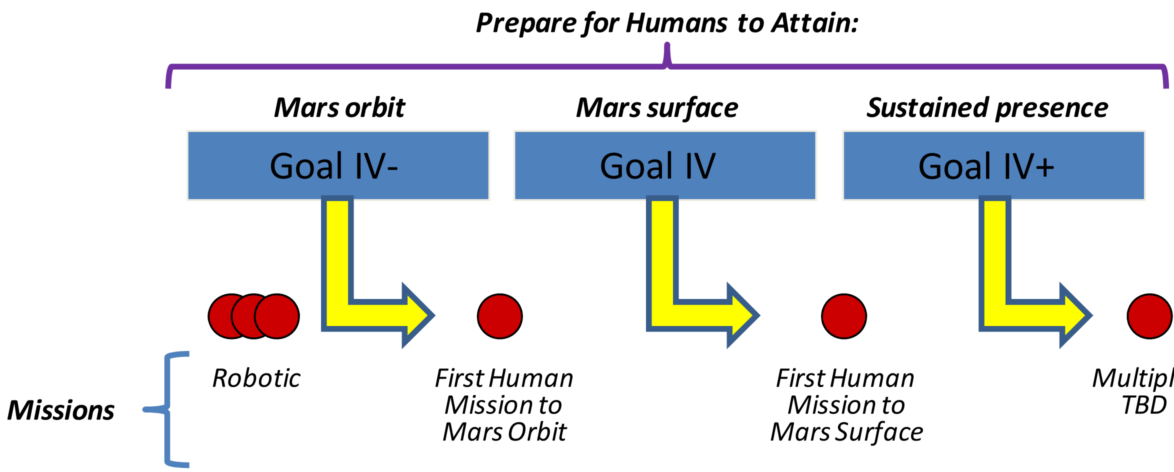


Figure 13-5 Goals for human presence in the Mars system.

Key Findings: The human exploration objectives for a Mars mission (including the MPD mission) include the following:

- Obtain knowledge of Mars, its moons, and the surrounding environment sufficient to design and implement human missions with acceptable cost, risk, and performance.
- Conduct technology, operations, and infrastructure demonstrations in transit to, in orbit around, or on the surface of Mars or Phobos and Deimos to reduce risk or cost for human missions.
- Incorporate partnerships (international, commercial, etc.) that broadens overall organizational participation but also lowers the total program cost for each partner.
- Incorporate multiple public engagement events spread across entire mission durations and using multiple media types.
- Prepare for sustained human presence.

As with the science objectives, the applications of teleoperations at Mars that will satisfy human exploration objectives are unclear at this time and require further study.

Table 13-3 identifies the key implementation implication for each of the exploration objectives.

Table 13-3 Implementation implications of exploration objectives.

EXPLORATION OBJECTIVES	IMPLEMENTATION IMPLICATION
Obtain knowledge of Mars, its moons, and the surrounding environment	a) Data to develop gravitational potential models for Phobos and Deimos; b) Imagery of TBD resolution with altimetry of the entire surfaces of Phobos and Deimos; c) Solar Particle Event (SPE) and GCR radiation measurements from orbit; d) Data to develop preliminary geological maps of Phobos and Deimos; e) Civil engineering data for safe landing and operations.
Conduct technology, operations, and infrastructure demonstrations	a) Exercise Mars surface sample return protocol; b) Collect system performance data (Environmental Control and Life Support System (ECLSS), power, etc.); c) Exercise independent crew operation procedures; d) Exercise orbital operations (e.g., rendezvous with a suitable target); e) Demonstrate In-Situ Resource Utilization (ISRU) on Phobos or Deimos.
Incorporate partnerships that broadens overall organizational participation	a) International Partner contribution of mission elements, experiments, and other equipment; b) Use of commercially available elements with (potential augmentation) to meet mission requirements; c) Include partnerships as applicable with other US government agencies.
Incorporate multiple public engagement events	a) Perform an early mission to the Martian system to engage the public and maintain interest in a Mars surface mission; b) Include student – developed experiments and projects (allocate time, mass, power, etc.), c) Include time in scheduled crew activities for public outreach activities during all mission phases.
Prepare for sustained human presence	a) Catalog elements and minerals types and concentrations on the surface and subsurface of Phobos and Deimos; b) Surface and near subsurface "civil engineering" properties at Phobos and Deimos; c) Long duration Mars atmospheric observations; d) Demonstrate ISRU processes for applicable mineral types; e) Demonstrate key elements (TBD) of long term orbital infrastructure.

13.1.1.5.3. Destination Activity Implementation Strategy

Team: Kevin Earle (Lead), Jeff Antol, Deborah Bass, David Coan, Kevin Daugherty, Mike Hembree, Sharon Jefferies, Ruthan Lewis, and David Reeves

Charter: Determine whether a worthwhile human mission to Phobos and Deimos can be accomplished with a high degree of confidence during an opposition-class (short-stay) mission opportunity. Determine operational timeline and required equipment, and formulate telerobotic operations (moons and Mars surface) and extravehicular activity (EVA) support strategies.

Key Findings: Based on the science and exploration objectives identified, preliminary results indicate that an opposition-class mission to Phobos and/or Deimos appears feasible. All currently identified science and exploration objectives could be accomplished in 56 days. The development of a conservative plan provides substantial schedule margin. Further studies are needed to optimize mission planning, understand implications of in-system teleoperations, and refine objectives definition. The preliminary DMC development approach utilized by the team is shown in Figure 13-6, and the high-level concept of operations (ConOps) developed is shown Figure 13-7. It should be emphasized that this preliminary. The ConOps represents a conservative existence proof and attempts were not made to optimize it.

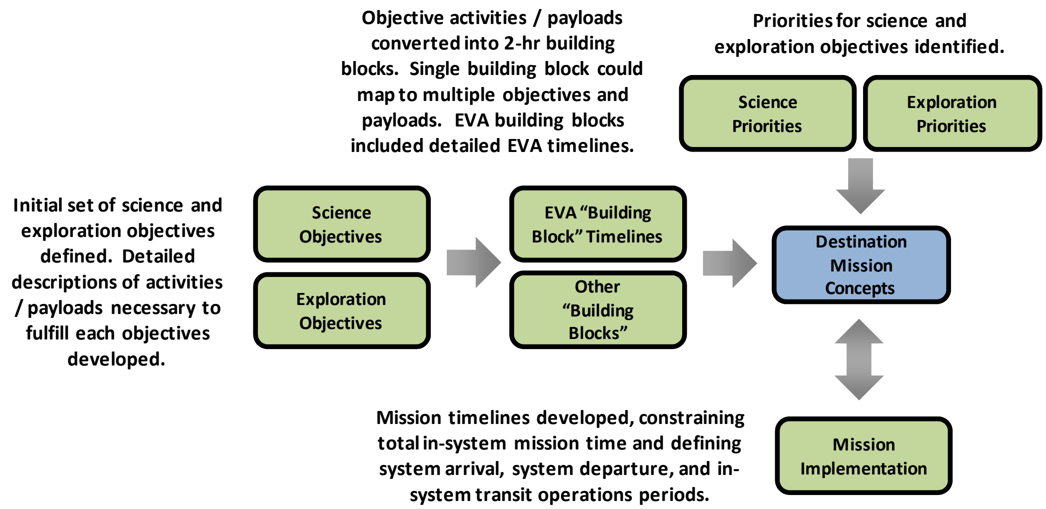
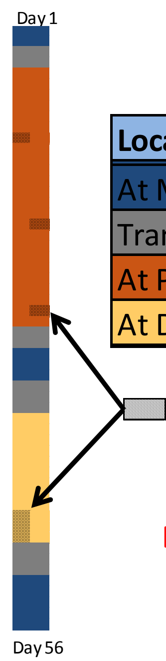


Figure 13-6 Preliminary DMC development approach.

Activity	Days	Total
Arrival Operations	2	2
Mothership to Phobos	2	4
Phobos Locations 1 & 2	8	12
Phobos Locations 4 & 6	8	20
Phobos Locations 3 & 5	8	28
Phobos to Mothership	2	30
At Mothership	3	33
Mothership to Deimos	3	36
Deimos Sites 1 & 3	7	43
Deimos Location 2 & TOA	5	48
Deimos to Mothership	3	51
Departure Operations	2	53
Contingency	3	56



Location	Days
At Mothership	10
Transfer	10
At Phobos	24
At Deimos	12

Denotes days where at least 1 SEV is at a location with no planned activities

Existence Proof

Figure 13-7 High-level concept of operations.

Allocations/resources for in-system telerobotics are not included in the preliminary MPD concept of operations because the objectives identified by the Science and Exploration Objectives and Requirements formulation teams could be achieved more efficiently with other means. The team did explore the potentials benefits and challenges of moving some of the robotic support team in-system (as shown in Figure 13-8) to achieve more decision-points per sol by reducing the communication latency: With refined or alternate objectives, in-system telerobotics would need to be reassessed.

Benefits:

- Increased situational awareness may reduce risk for more challenging operations.
- Progress of activities can increase due to multiple decision points per sol.
- Use of unconventional scientific platforms (e.g., airplanes and hoppers).

- Transient science acquisition (e.g., dust devils on Mars, meteorite impact on Phobos/Deimos).

Challenges:
- Much higher operations cost due to large engineering and operations support staff required for crew support.
- In-system mission periods have limited durations, much shorter than durations available for Earth-controlled operations.
- Time availability of in-system crew; crew has many other activities that they need to perform.
- Additional crew training requirements for telerobotic operations.

Unresolved Issues and Forward Work:

- Derive payload masses associated with performing destination activities and aggregate to determine outbound and inbound mission requirements.
- Optimize operations to align activity order with objective priority.
- Examine implications of shorter duration mission concepts (e.g., 30 days).
- Investigate alternative EVA operational modes, such as multiple crew members on EVA simultaneously, use of telerobotics in close proximity to crew, and alternative system hardware (e.g., advanced EVA maneuvering unit).
- Align systems performance assumptions with architecture design.
- Assess implications for incorporating in-system telerobotics.
- Determine required contingency duration between return from last moon excursion and Mars system departure.
- Investigate hybrid control approach for telerobotics to best leverage advantages of in-system vs. Earth-based locations (e.g., priority-based plan from Earth with crew monitoring and switching based on real-time observations) and enable rapid crew intervention to avoid damage or loss of robotic asset(s).

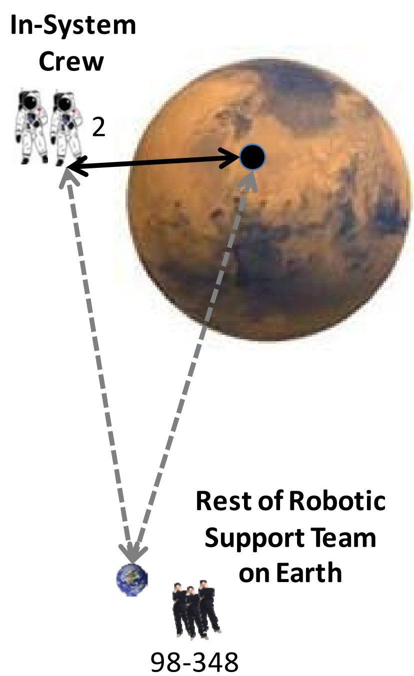


Figure 13-8 In-system telerobotics approach.

13.1.1.5.4. Mission Implementation Strategy

Team: Bret Drake (Lead) and Brent Barbee

Charter: Formulate the overall MPD mission exploration strategy options and provide a preliminary analysis of trajectories within the Martian system to facilitate the destination activities.

A trip to Mars with a return back to Earth is a double rendezvous problem flown in heliocentric space. The first rendezvous outbound, with Mars, must be dealt with considering its influence on the second rendezvous, inbound with Earth. Practical considerations dictate favorable, and different, planetary alignments relative to the sun for outbound and inbound transfers. These considerations result in two distinct mission classes: short-stay class missions and long-stay class missions. Short-stay class missions (see Figure 13-9) are characterized by relatively short periods spent in the vicinity of Mars (generally 30-60 days). As such, these missions will tend to be highly scripted with pre-planned operational timelines. Due to the short time at Mars, there will be less time available for mission re-planning due to contingencies or large unanticipated discoveries. Long-stay class missions (see Figure 13-10) are characterized by long periods spent in the vicinity of Mars (330-560 days) and overall long mission durations (900+ days). These long-stay missions provide ample time for re-planning mission operations. It is envisioned that upon arrival at Mars, a very pre-planned scripted operational scenario will be followed. As the mission evolves, a more free-flowing collaborative (with Earth) scenario would follow.

Human Research Program Inputs: NASA's Human Research Program (HRP) and medical support for the crew are not synonymous. Medical support is focused on individual health during a mission while HRP also considers mission performance and post mission health. There can be significant overlap in the kinds of data and physical samples collected as well as on-board analyses made.

Based on our current understanding of the human body's reaction and adaptation to micro-gravity, medical interventions to maintain health (e.g., exercise) are expected to be qualitatively the same for Mars long-stay surface mission transits (6 months in micro-g), longer duration NEA missions (up to ~12 months in micro-g), or Phobos-Deimos missions (500 – 1000 days in micro-g). Data and samples collected for HRP-related research are expected to be the same types, with the quantities driven by mission duration. Our current understanding of human psychology and the impact of extended duration confinement is incomplete, and additional research is expected to require additional understanding of the impact of extended confinement (up to ~12 month NEA missions and ~1000 day Phobos-Deimos mission) on individuals and crews within deep space habitats before any conclusions can be made. Crew exposure to the radiation environment of deep space remains a key HRP risk area. Minimizing crew exposure to radiation (GCR and SPE) is a key mitigation strategy (e.g.., reduce total mission duration for orbital missions).

Crewed Mission Transportation and Exploration Systems

Figure 13-11and Figure 13-12 provide preliminary estimates of the expected Initial Mass in Low-Earth Orbit (IMLEO) for the crewed transportation architectures high thrust and low thrust propulsion approaches for a mission to the Mars system respectively. For example, total architecture mass estimates using nuclear thermal Propulsion (NTP) for a 550-day opposition-class mission range from 350-1000+ tons (opportunity dependent). These estimates exclude destination systems, which will likely be pre-deployed to the Martian system. Long-stay (conjunction-class) missions offer the advantage of lower overall mission mass (due to lower total ΔVs) and longer time in the Mars system for exploration activities, but with a longer overall mission time. Additional factors (e.g., cost, risk, mission operations, and value of additional science/exploration time) must be taken into account before reaching a conclusion on the most appropriate mission mode to achieve mission objectives. The following element assumptions were used to develop these estimates:

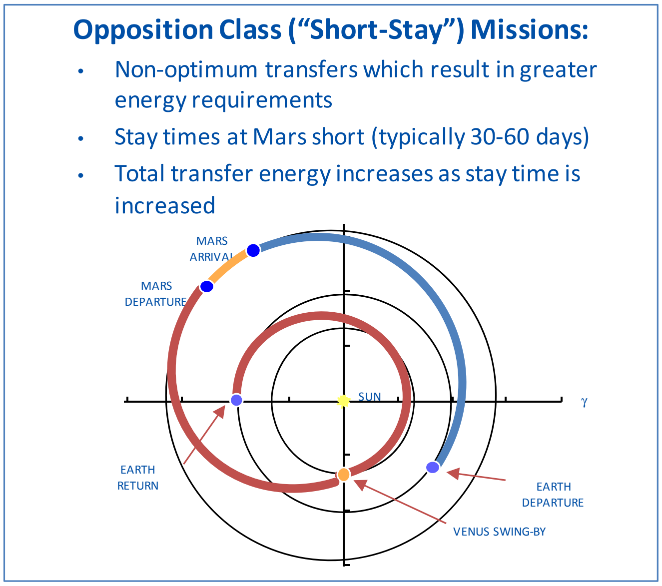


Figure 13-9 Short-stay class mission.

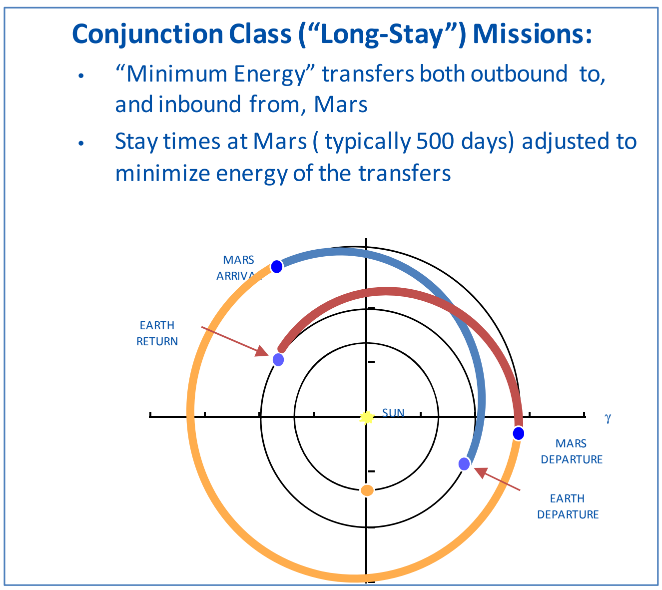


Figure 13-10 Long-stay class mission.

Multi-Purpose Crew Vehicle:

- Consistent with HAT
- CM inert = 9.8 t
- SM inert = 4.5 t
- SM specific impulse = 328 s

Deep Space Habitat

- Sizing consistent with HAT Cycle-C
- Mass Range : 28-65 t
- Consumables loaded based on crew size & mission duration

Chemical Propulsion Stage

- Sizing consistent with HAT Cycle-C
- Parametric design with each stage optimized
- Zero-boiloff cryo management
- Stage fraction ~ 23%
- Specific impulse = 465 s

Solar Electric Propulsion

- Consistent with HAT
- Spacecraft alpha ~30 kg/kW
- Specific impulse = 1800-6000 s
- Xe tank fraction = 5%
- Total power varies

Nuclear Electric Propulsion

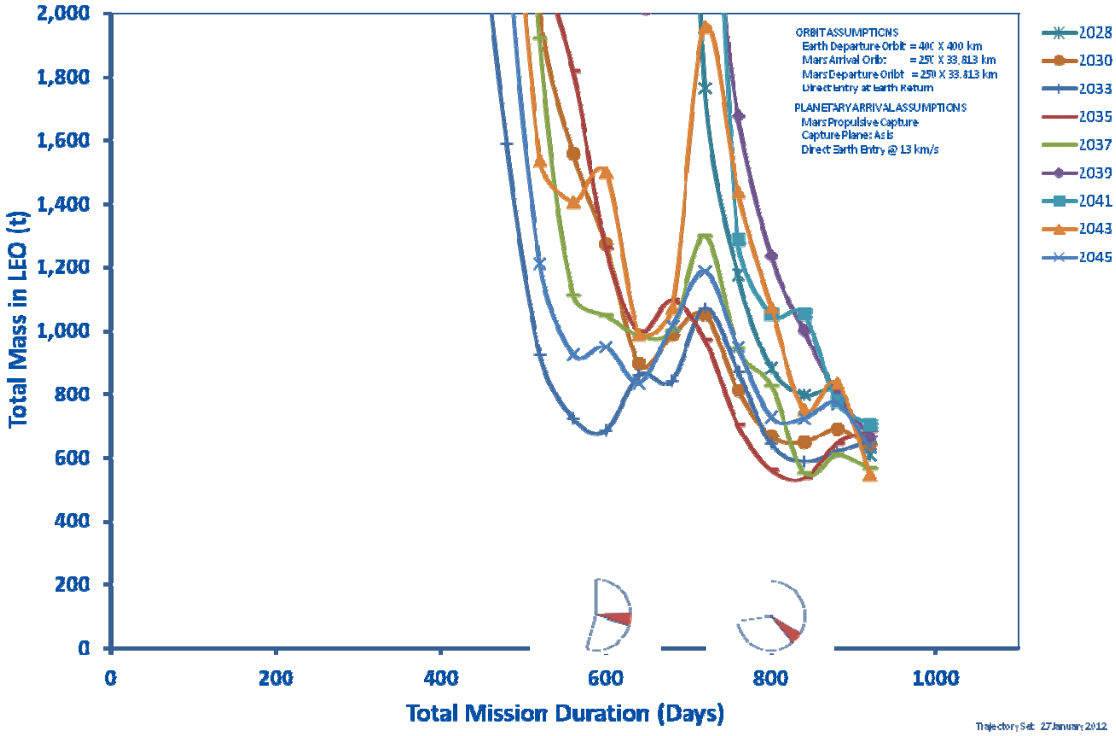
- Spacecraft alpha ~20 kg/kW
- Specific impulse = 1800-6000 s
- Xe tank fraction = 5%
- Total power varies

Nuclear Thermal Propulsion

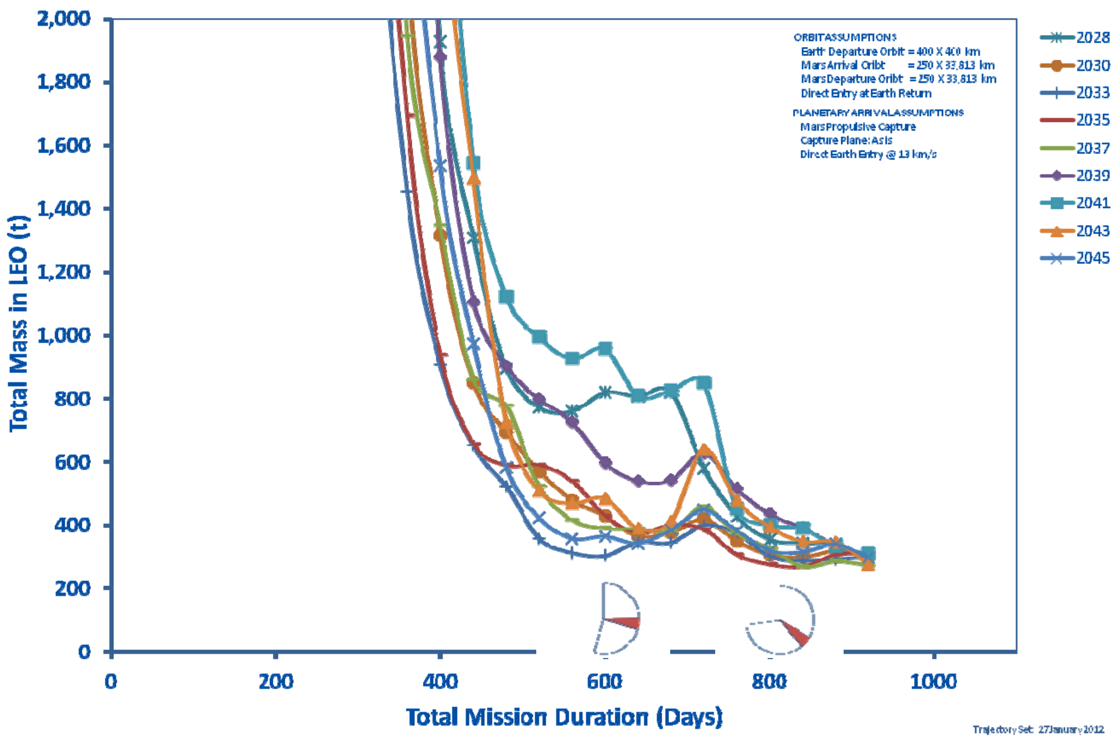
- Consistent with Mars DRA 5
- NERVA-derived common core propulsion (20 t core)
- 3 x 111 kN engines
- Specific Impulse = 900 s
- All LH2 fuel with zero boil-off
- Drop tanks @ 27% tank fraction

Space Launch System
Gross Performance ~ 130 t

- Net Performance ~ 120.4 t (HAT assumptions for reserve and adapters)
- Performance estimates to negative perigee conditions: (-87 km x 241 km)

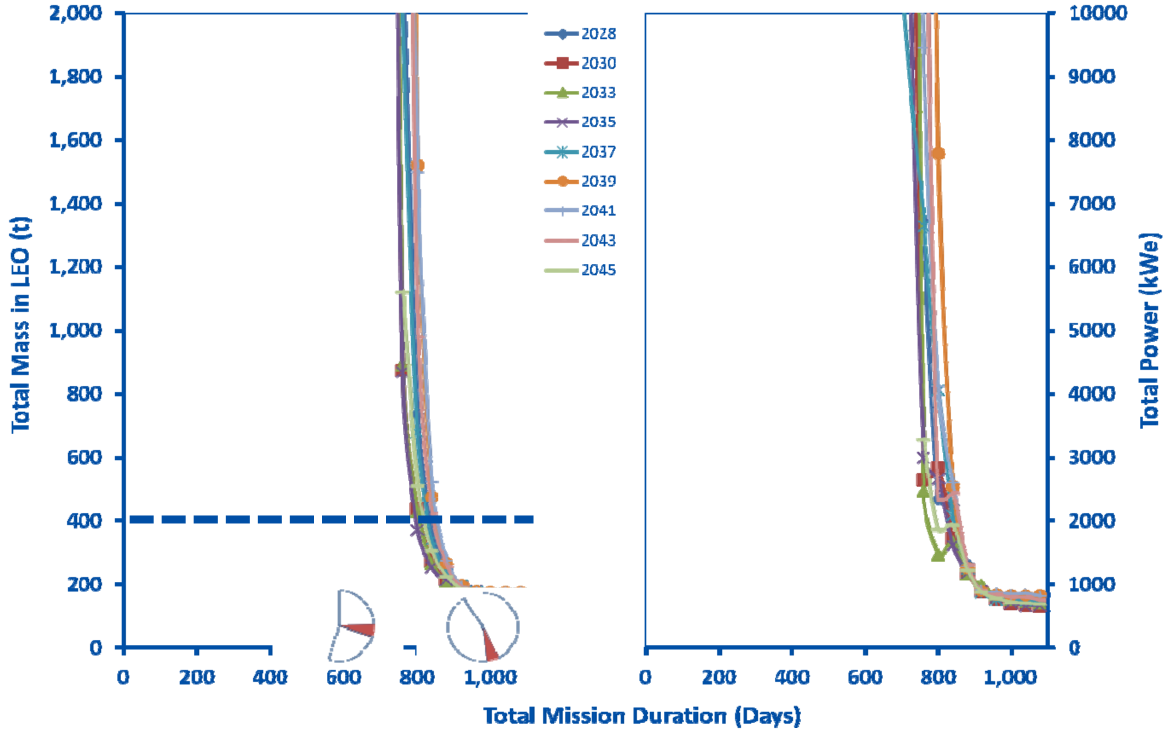


(a) Chemical propulsion

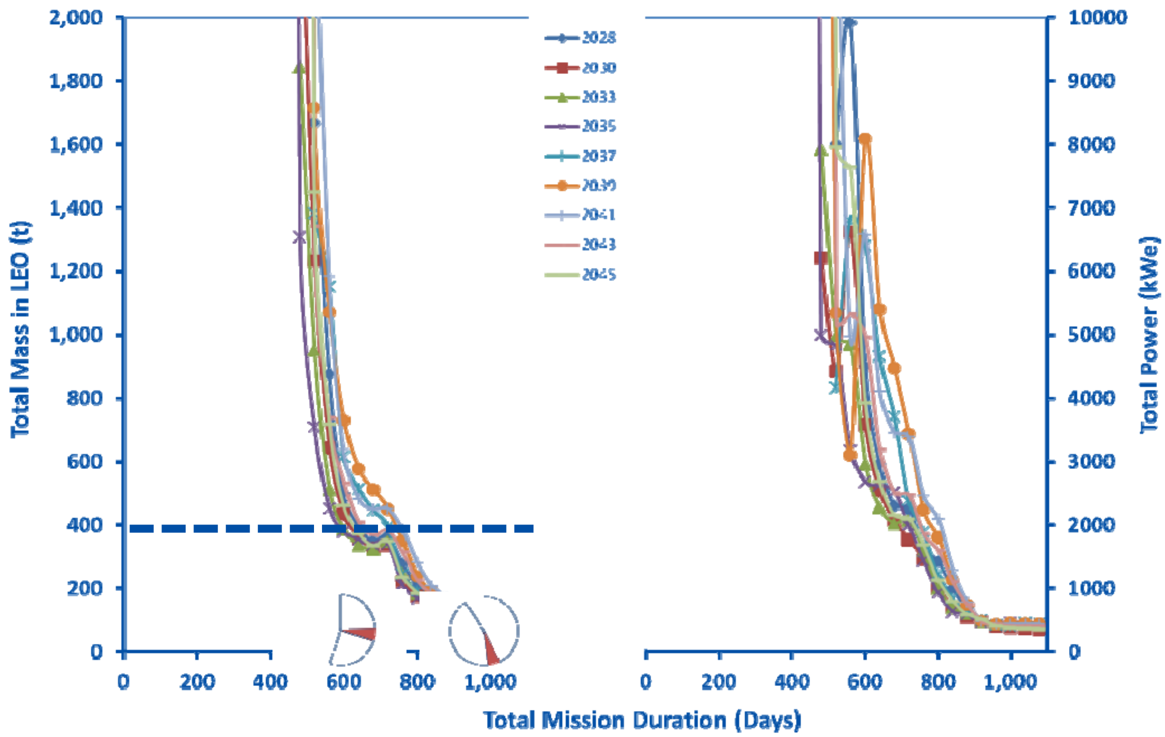


(b) Nuclear thermal propulsion

Figure 13-11 Total architecture mass as a function of total mission duration for high thrust propulsion concepts.



(a) Solar electric propulsion



(b) Nuclear electric propulsion

Figure 13-12 Total architecture mass as a function of total mission duration for low thrust propulsion concepts.

Mars Orbit Capture and Departure Dynamics

Both Phobos and Deimos are essentially in the equatorial plane of Mars, with nearly circular orbits at 9,378 km and 23,459 km, respectively. Earth-Mars trajectory arrival and departure geometries are not in the equatorial plane, thus additional orbital maneuvers (inclination change and orbit lower/raise) are required once the necessary crew parking orbit is established. For high thrust approaches, a multi-burn strategy used to account for planar alignments and capture time is short (hours to days) depending on parking orbit chosen. For low thrust approaches, the plane adjustments are made at Mars arrival (sphere of influence) and capture duration is long (weeks to months). The duration depends on the parking orbit along with the power, thrust, and specific impulse of the low-thrust propulsion system.

Key Findings: Human missions to the moons of Mars are conducted entirely in deep-space, and reducing the exposure of the mission crew to the hazards of deep-space is of prime concern. Practical considerations (e.g., transportation technology and number of launches) will limit mission durations to not much less than 600 days, and thus, human health issues cannot be obviated by propulsion technology alone. If there is no significant difference between 600 and 900 days from human health or overall mission risk and operations perspectives, then long-stay (conjunction-class) missions offer the advantage of lower overall mission mass and longer time in the Mars system for exploration activities. However, other factors (e.g., cost, risk, and value of additional science/exploration time) must be taken into account before reaching a conclusion on the most appropriate mission mode to achieve mission objectives.

13.1.1.5.5. Synergies with Cis-lunar Activities

Team: Mark Lupisella (Lead), Jeff Antol, Deborah Bass, Dave Beaty, Kevin Daugherty, Lee Graham, Ruthan Lewis, and Dan Mazanek

Charter: Identify how cis-lunar space missions and activities can provide preparation for an MPD mission and how an MPD mission might enhance cis-lunar activities.

The following are the potential areas for synergy with cis-lunar activities identified by the team:

- Human Research (e.g., radiation effects and mitigation)
- Telerobotics (e.g., low latency surface telerobotics)
- Mission Systems and Support (e.g., system reliability and logistics)
- Long-term Deep Space Human Operations (e.g., crew autonomy)
- Proximity Operations (e.g., crew mobility, worksite stabilization)
- Sample Return (e.g., return samples to cis-lunar facility)
- Forging Partnerships (e.g., crew telerobotically control partner surface asset)
- Public Engagement (e.g., test crew activities for public outreach with delay)

High and medium priorities for cis-lunar synergy are shown in Table 13-4. The activities were prioritized based on the following criteria:

1. Objective alignment
2. High potential, but high uncertainty – suggesting need for in-space tests
3. Feasibility

Low-Latency telerobotics operation may be a useful strategy, particularly if human missions stay out of gravity wells for some time. As the ultimate value of telerobotic science on Martian surface is yet to be determined, testing in cis-lunar space to explore potential value and to test systems is probably worthwhile. The following levels of telerobotic science, along with the team's assessment of their probability of being implements, are:

- Operations and navigation: achievable
- Basic science (e.g., instrument positioning, sample acquisition): probably achievable
- Detailed science measurements and interpretation (most challenging part): perhaps partially achievable,

needs testing

The potential difficulty of telerobotic science noted above suggests the possibility of parallel operational paradigm that includes two parallel paths, somewhat analogous to tactical vs. strategic planning, or short- vs. longer-term planning: 1.) telerobotic science; and 2.) "back room" science using Earth-based support.

Table 13-4 High and medium MPD Cis-lunar synergies priorities.

Synergy Activity	Priority
HUMAN RESEARCH: all activities (except artificial gravity**)**	High
TELEROBOTICS	
Simulate delays and different orbital operational implications for Mars surface	High
Conduct "fast" traverses to assess potential science return (could help with diversity)	High
Assess real-time science responsiveness	High
Perform analog tests for telerobotic operations of MPD surfaces – relates to proximity operations synergy. *An effective precursor mission could substantially reduce (but not eliminate) the dependency on telerobotic surface interaction.	High*
Conduct public outreach activities	Med
MISSION SYSTEM AND SUPPORT	
Radiation shielding	High
Life support system reliability	High
Medical support: health monitoring/treatment, including for planetary protection purposes	High
Subsystem serviceability and sparing	Med
Test pre-deploy strategies - e.g., consumables, fuel, and Automated Rendezvous and Docking (AR&D)	Med
LONG-TERM DEEP SPACE HUMAN OPERATIONS	
Crew autonomy / control authority tests	High
Verify & mature long-duration crew medical care operations	High
PROXIMITY OPERATIONS: Crew translation, restraint, worksite stabilization (build crude analog)	Med
SAMPLE RETURN: Return samples from lunar orbit/surface to cis-lunar asset as analog to returning samples from Martian orbit/surface to return vehicle	Med

Key Findings: There are a number of promising activities to conduct in cis-lunar space to help prepare for a human MPD mission. Most human research needed for a MPD mission can be conducted during cis-lunar missions. Crew autonomy is a key area to test during cis-lunar missions. Telerobotics has high potential, but also high uncertainty for science effectiveness and requires additional analysis and testing. Finally, large amounts of sample, that may not be returned directly to Earth, could be received at a cis-lunar facility.

13.1.1.5.6. Synergies with Human and Robotic Precursor NEA Missions

Team: Paul Abell (Lead), Julie Castillo-Rogez and Dan Mazanek

Charter: Identify synergies between human and robotic NEA missions and Phobos/Deimos missions. Determine the information and experience that can be gained from NEA missions prior to Mars system missions and assess the associated advantages.

NEAs and Phobos/Deimos are small airless bodies with similar physical characteristics, but represent distinct and separate destinations for robotic and human exploration. NEAs are any asteroid passing within 1.3 Astronomical Units (AU) of the Sun, while Phobos and Deimos are natural satellites of Mars at ~1.52 AU. The two NEAs that have been visited by robotic spacecraft, (433) Eros and (25143) Itokawa are shown in comparison to Phobos and Deimos in Figure 13-13.

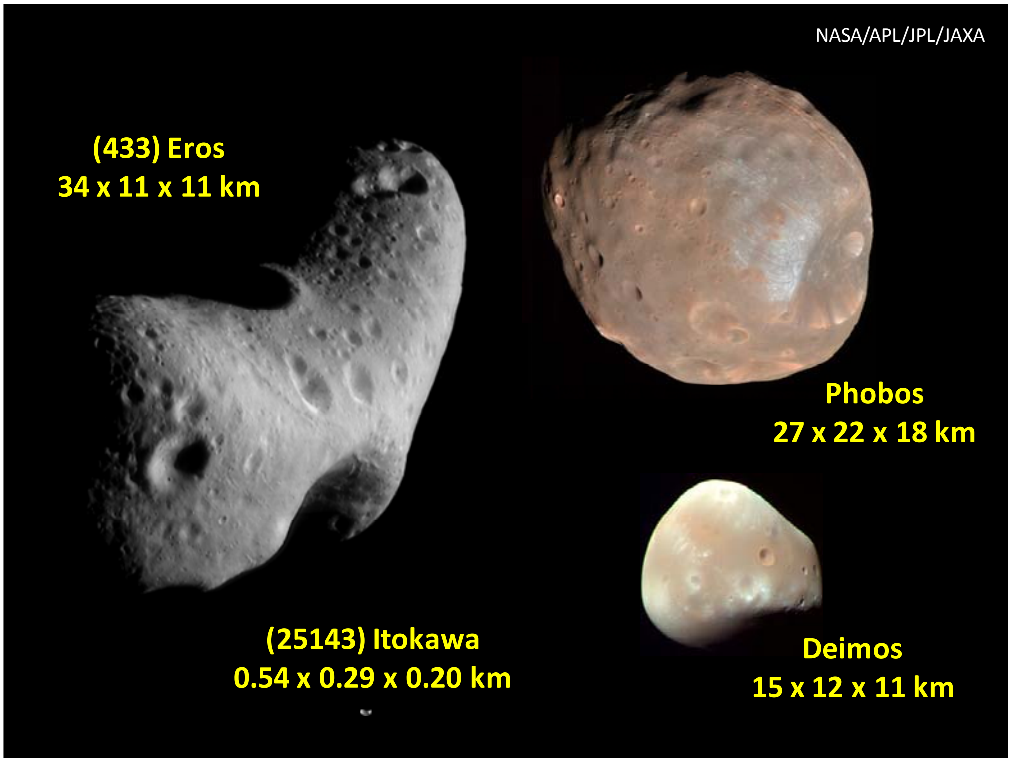


Figure 13-13 Robotically visited NEAs at approximate scale with Mars moons.

There are a number of synergies between missions to NEAs and missions to Phobos and Deimos. These synergies exist during the proximity operations, while conducting surface operations at the target, and even during the transit to/from the target (NEA or Phobos/Deimos).

Robotic Precursor Activities: Enable target identification and selection for future human mission activities, constrain internal and near-surface structure on regional and global scales, and characterize basic physical properties relevant for science and future human safety, performance, and operations.

Human Exploration/Operations: Provide lessons learned from building reliable power, propulsion, communication, and life support systems for long duration (> 30 days) missions and allow a better understanding of how to operate in close proximity to, and at the surface, of a non-cooperative object in a low gravity regime.

Small Body Science: Aids understanding of the creation of our solar system since small bodies are the left over primitive materials from the earliest stages of solar system formation (e.g., potential Phobos/Deimos asteroid connections). Increased knowledge of these objects physical characteristics and their constituents also helps to refine models for the delivery of materials (organics, volatiles, water, etc.) that may have been instrumental for the formation of the early Earth and evolution of life.

In Situ Science Combined with Sample Return: Enables better understanding of these bodies' origin/dynamical history, nature of their material composition, thermal properties, oxidation state, and collisional histories. Evidence from the meteorite record and remote sensing observations of NEAs and Phobos/Deimos (both from ground-based and space-based assets) suggest that some of these objects contain significant amounts of resources (water and precious metals). Hence resource utilization synergies exist by performing extraction demonstrations of small token quantities as a proof of concept (i.e., extraction of volatiles from NEA or Phobos/Deimos materials) and evaluating the effectiveness of using these resources for life support, propulsion, and other potential applications to enhance safety and efficacy of human spaceflight.

Transit Science: Enhances the science return of the mission and provides multiple opportunities for science to be conducted en route to and from primary target (e.g., planetary science, life science, astrophysics, heliophysics). These types of observations and experiments can be performed for missions to either NEAs or Phobos/Deimos.

Human Factors: is an area of study that is crucial for enabling human space flight to destinations beyond the Earth-Moon system. Many human factors are common and relevant to both NEA and Mars moon missions. Researcher could measure effects of communication delays/blackouts and impact to crew morale/performance, characterize synergistic effects of radiation, microgravity, crew confinement, on the human immune system during extended duration deep space voyages, and monitor psychological effects of living in deep space for extended periods of time with no rapid return possibilities. Such a wealth of data would aid in better designs for improved spacecraft operations/performance and mitigate the effects on the deep space environment on human physiology and psychology.

Key Findings: There are numerous synergies between human missions to NEAs and missions to Phobos and Deimos. NEA missions can provide the opportunity to become proficient with human operations around a non-cooperative object in a low gravity regime. These synergies will be important for any Human mission to the Martian moons, but will be particularly relevant if short-stay missions are conducted due to the constrained duration for operations within the Martian system.

13.1.1.5.7. Robotic Precursor Requirements for a Human Mission to Mars Orbit and its Moon

Team: Paul Abell (Lead), Deborah Bass, Dave Beaty, Tony Colaprete, Dan Mazanek, along with additional team members Jim Head (Brown), Scott Murchie (APL) and Andy Rivkin (APL)

Charter: Identify the strategic knowledge gaps (SKGs) and required robotic precursor measurements necessary to help inform a short-stay human orbital mission to interact with Phobos and Deimos. This study area was included to support of Mars Exploration Program Analysis Group (MEPAG) Precursor Science Analysis Group (P-SAG) Sub-Team #6.

A robotic precursor to Phobos and Deimos could provide significant risk reduction by addressing strategic knowledge gaps early enough to inform human mission design. There are two areas of SKGs. The first area addresses SKGs related to aspects of the orbital mission and the second area addresses SKGs associated with visiting the Martian moons. In order to adequately explore the Martian moons the SKGs related to the orbital only mission should also be included for consideration. The SKGs related to the orbital aspects of the mission along with the mission relevant parameters and the team's priority are provided in Table 13-5 and those associated with visiting Phobos and Deimos are provided in Table 13-6.

Table 13-5. SKGs related to an orbital mission

Strategic Knowledge Gap	Human Mission Relevant Measurements	Priority
Atmosphere properties related to aerobraking /aerocapture	Temperature, winds, aerosol abundance and profile; global and diurnal coverage	Medium-High
Particulate environment	Spatial variation in size-frequency distribution of Phobos/Deimos ejecta particles in Mars orbit	Medium

Table 13-6 SKGs related to a mission to Phobos or Deimos.

Strategic Knowledge Gap	Human Mission Relevant Measurements	Priority
Mineralogical & chemical composition	Elemental / chemical composition; spatial distribution of major geologic units; ISRU potential	High
Regolith mechanical & geotechnical properties	Size-frequency distribution; density, compressibility, adhesion; spatial variation in thickness/properties	High
Gravitational field	Spherical harmonic terms of moons' gravitational fields	Medium
Electrostatic charging & plasma fields	Electric fields in proximity to surface, plasma emanating from surface	Low
Thermal environment	Temperature variation diurnally and with depth	Low
Radiation environment	Local radiation environment (including secondary radiation) near the Martian moons	Undetermined

Mars Orbit Insertion (MOI). Since aerobraking/aerocapture using the Martian atmosphere may be pursued for orbital missions, the knowledge necessary to enable aerobraking of large masses (> 10 t) should be obtained prior to a crewed mission. More knowledge of the Martian atmospheric properties may allow aerobraking to be developed, which may enable more massive robotic and human missions to be conducted to the Martian system earlier than presently anticipated. . The data relevant for making an informed decision on the implementation of aerobraking/aerocapture would significantly influence the mission architecture and resulting mass required.

Orbital Debris. Since the orbital particulate environment is potentially more significant in high Mars orbit (near Phobos/Deimos and in the equatorial plane) than in low Mars orbit, direct measurement of the debris flux should be obtained. This is important since, under current architectures, spacecraft with significant cross-sectional areas may spend significant time in the equatorial plane and make repeated passes through this region of the Martian system.

Mineralogy and Chemical Compositions. A better understanding of the mineralogy/chemistry of Phobos and Deimos is needed to support productive science operations and may also influence operations planning. For example, detection of organics and volatile compounds on Phobos/Diemos will drive different science and exploration objectives than mineralogies that do not contain such compositions.

Regolith. Regolith contact measurement and mapping are needed for operations planning and surface interaction considerations in order to better define the equipment and instrumentation that will be most effective in achieving the exploration and scientific objectives of the Phobos/Deimos mission. Regolith characteristics on a local scale may affect the method and extent of surface interaction to be conducted by the crew and their exploration assets.

Gravity. Gravitational field measurements are recommended for planning proximity operations, including identification of station keeping modes. These data are relevant for optimizing propellant usage during proximity operations and understanding the effects long term variations of complex gravitational fields on assets in close proximity to one another.

Electrostatics. Electrostatic charging and plasma environment influences engineering of surface elements and EVA equipment designs. Such information is necessary to help inform better designs and countermeasures for the development of exploration operations and interactions of systems with these airless bodies.

Thermal. Thermal environmental conditions vary significantly over diurnal time scales and with regolith depth. These data would inform the design of surface and sub-surface elements, including EVA equipment and scientific instrument designs..

Radiation. The need for radiation measurements related to tissue equivalent response near the Martian moons (not to basic measurement of GCR and SPE) is still under debate. A better understanding of the interaction with Phobos/Deimos for shielding/secondary effects would be beneficial, but it is not clear that it is required.

Figure 13-14 depicts various precursor platform orbital options. An orbiter in low-Mars orbit (1) is well suited for addressing measurements associated with the Martian atmosphere. An orbiter in a high-Mars orbit or an elliptical orbit with both low- and high-aspects (2) would be able to address the natural debris environment in the equatorial plane and could collect partial information on temperature, mineralogy, and gravity of Phobos/Deimos. A precursor that performs a rendezvous and a landing with Phobos and/or Deimos (3) is required to fully address the SKGs at Phobos/Deimos. Rendezvous-only missions (4) and sample return missions (5) were considered by the team, but these missions either do not adequately address, or are not necessary for addressing, the relevant SKGs for future human interaction at these moons .

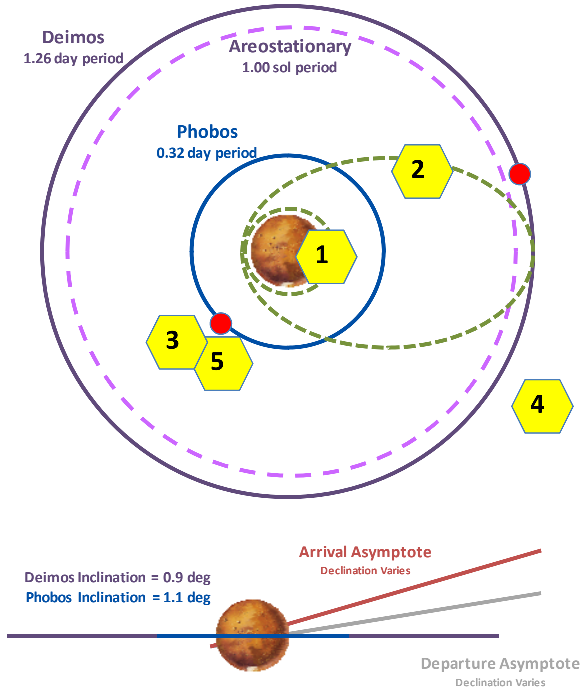


Figure 13-14 Precursor platforms.

Key Finding: A robotic precursor to Phobos and Deimos could provide significant risk reduction for a future human orbital Mars mission by addressing strategic knowledge gaps early enough to inform human mission design.

13.1.1.5.8. Activity Conclusions

Preliminary results from the MPD activity indicate that a meaningful human orbital mission to explore both Martian moons and retrieve a MSR cache from low Mars orbit could be performed during an opposition-class mission opportunity. The initial destination mission plan indicates that 56 days are required to accomplish all science and exploration objectives. Margin and mission reduction opportunities provide confidence that a successful and worthwhile mission could be completed within 60-90 days in the Mars system. Preliminary parametric based estimates of the expected initial mass in low-Earth orbit (IMLEO) for a transportation architecture utilizing nuclear thermal propulsion to support an opposition-class mission (total duration of approximately 550 days) range from 350 to over 1000 metric tons. The IMLEO is highly dependent on the Mars departure opportunity, with 2033 offering a minimum in the 2030-2040 timeframe. Detailed mass sizing and volumetric analyses are needed to validate these initial estimates. Finally, the results from each of the activity study areas provide valuable information regarding the development of a human MPD mission and the synergistic activities required prior to undertaking such an exploration endeavor.

13.1.1.5.9. Summary

Through a comprehensive approach starting with the development of key mission objectives and working through activity implementations and overall mission implementation strategies, our preliminary results suggest that an opposition-class mission to Phobos and Deimos could meet the identified objectives. In addition, there are key synergies to leverage with human missions to cis-lunar destinations and NEAs. Robotic precursor missions to Phobos and Deimos would provide significant risk reduction by addressing strategic knowledge gaps early enough to inform human mission design.

13.1.1.5.10. MPD DMC Primary Team Members:

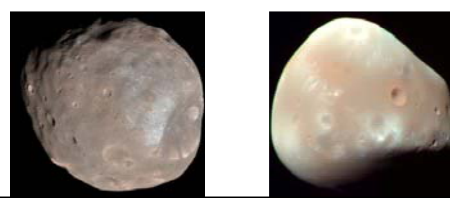
Paul Abell (JSC), Jeff Antol (LaRC), Brent Barbee (GSFC), David Beaty (JPL), Deborah Bass (JPL), Julie Castillo-Rogez (JPL), David Coan (JSC), Tony Colaprete (ARC), Kevin Daugherty (LaRC), Bret Drake (JSC), Kevin Earle (LaRC), Lee Graham (JSC), Mike Hembree (JSC), Steve Hoffman (JSC), Sharon Jefferies (LaRC), Ruthan Lewis (GSFC), Mark Lupisella (GSFC), Dan Mazanek (LaRC – Study Lead) and David Reeves (LaRC)

13.1.2. Mission Design for the Exploration of Phobos and Deimos

Primary Contributors:
Brent W. Barbee, NASA Goddard Space Flight Center, USA
Damon Landau, Ph.D., Jet Propulsion Laboratory, California Institute of Technology, USA

The two moons of Mars, Phobos and Deimos, are among the potential destinations currently under consideration by NASA for future human exploration missions. This section presents results from NASA's Mars-Phobos-Deimos Working Group study in the areas of orbit analysis and trajectory design for human space flight missions to explore Phobos and Deimos. The evolution of the orbits of the moons under natural perturbations are analyzed in this section, which informs the design of trajectories to arrive at Mars in a highly elliptical orbit, rendezvous with each moon in turn, and then depart Mars. The abilities of each moon to support captured orbits during proximity operations are also considered.

The two moons of Mars, Phobos and Deimos, are among the potential destinations currently under consideration by The National Aeronautics and Space Administration (NASA) for future human exploration missions. Figure 13-15 shows Phobos and Deimos as seen by the Mars Reconnaissance orbiter in 2008, along with a few key physical properties of the moons.



Characteristic	Phobos	Deimos
Mass (kg)	1.08×10^{16}	1.80×10^{15}
Dimensions (km)	26.2 x 22.2 x 18.6	15.6 x 12.0 x 10.2
Rotation Period	Synchronous	Synchronous

Figure 13-15 Characteristics of Phobos and Deimos

NASA formed the Mars-Phobos-Deimos Working Group in early 2012 to study the design of missions to explore Phobos and Deimos. While the study spanned many aspects of the overall problem, this section is focused on the design of trajectories to arrive at Mars, rendezvous with each of the moons, and then depart Mars for Earth return. First analysis of the evolution of each moon's orbit under the influence of natural perturbations was conducted and then utilized to design sequences of maneuvers for exploring the moons. The maneuver sequences account for the conditions at Mars arrival on the incoming hyperbolic approach trajectory from interplanetary space, as well as the conditions that must obtain at Mars departure for injection into the outbound hyperbola leaving Mars. Also considered is both the phasing and relative motion required for rendezvous trajectories between the moons, as well as each moon's ability to support captured orbits during proximity operations.

13.1.2.1. The Orbits of Phobos and Deimos

In this study the precise orbits of Phobos and Deimos were considered under the in influence of natural perturbations including non-spherical Mars gravity and the gravity of other significant solar system bodies. The interval for analyzing the evolution of the moons' orbits was January 1, 2030 through January 1, 2040. Precision ephemeris files for each moon were downloaded from the NASA Jet Propulsion Laboratory (JPL) HORIZONS system[1] to study the areocentric inertial motion of each moon. Figure 13-16 provides a three-dimensional perspective view of the nominal areocentric orbits of Phobos and Deimos.

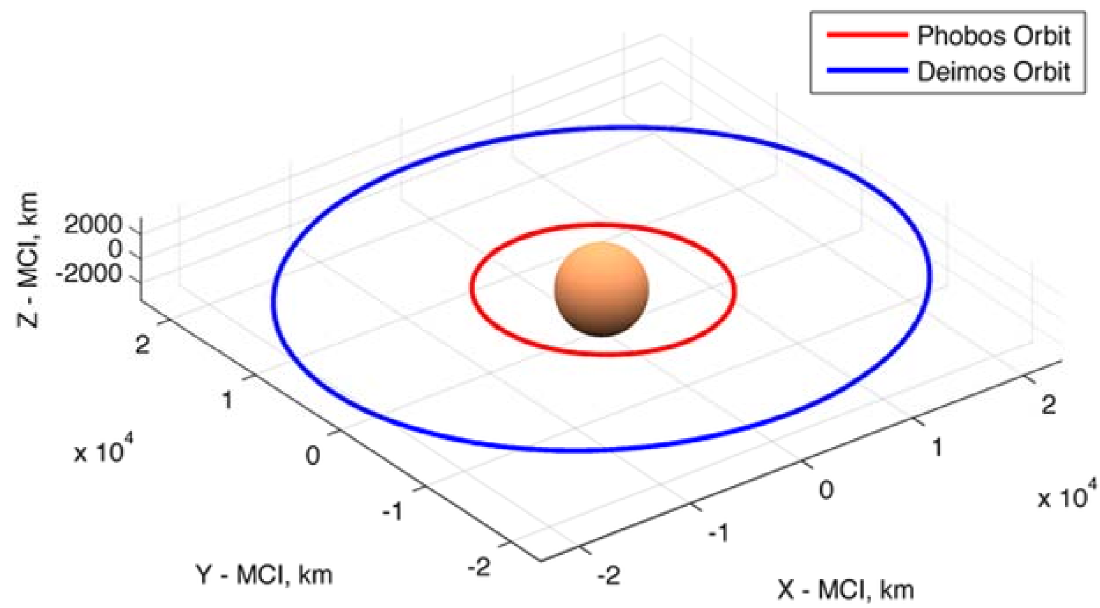


Figure 13-16 Nominal areocentric inertial orbits of Phobos and Deimos

At first glance the orbits of the orbits of the moons do appear circular and coplanar, with Phobos in a much lower orbit than Deimos. However, closer examination of the ephemerides reveals the exact nature of the orbits. Figure 13-17(a) shows the semi-major axes of the orbits as functions of time, and they appear quite constant throughout the analysis interval. However, while not shown here, zooming in on the semi-major axis data reveals small but bounded fluctuations in each orbit's semi-major axis. These slight fluctuations in semi-major axis manifest later when the synodic[2] period between the moons is considered.

Figure 13-17(b) presents the eccentricity of each orbit as a function of time. Although the eccentricity of the orbits is relatively small, neither eccentricity is negligible nor is Phobos' orbit much more eccentric than that of Deimos. Furthermore, the amplitude of the fluctuations in Phobos' orbit eccentricity is larger than the amplitude of the fluctuations in Deimos' orbit eccentricity; this is likely due to the fact that Phobos is in a much lower orbit than Deimos and therefore more strongly perturbed by Mars' gravitational field. However, the eccentricity of Phobos' orbit does remain bounded throughout the analysis interval despite its short period fluctuations.

Figure 13-17(c) shows the inclination of each orbit as a function of time. While both inclinations are low, neither

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1 http://ssd.jpl.nasa.gov/?horizons

2 The synodic period is the time required for any phase angle to repeat itself. Proper phase angle is needed to minimize the total ΔV required for the rendezvous sequence.

orbit is equatorial. Phobos' orbit inclination is more stable than that of Deimos, exhibiting no discernible secular variations. Deimos' orbit inclination, on the other hand, is steadily trending downward. Both orbits exhibit small short period inclination variations.

Figure 13-17(d) shows the right ascension of the ascending node (RAAN) of each orbit as a function of time. The RAAN of Deimos' orbit is gently, but steadily, trending downward throughout the analysis interval, while the RAAN of Phobos' orbit is precessing rapidly (changing through 2π radians every 2 years or so) under the influence of Mars' non-spherical gravitational field.

Figure 13-17(e) shows the argument of periarieon of each orbit as a function of time. The overall character of the argument periarieon evolution is similar to that of the RAAN evolution discussed previously. The argument of periarieon of Deimos' orbit trends upward gently but steadily throughout the analysis interval and also experiences noticeable short period variations. The argument of periarieon of Phobos' orbit precesses rapidly, changing through 2π radians every 1 year or so (approximately twice as fast Phobos' RAAN change) and in the opposite direction of Phobo's RAAN. Those results are consistent with the analytical treatment of changes in RAAN and argument of periarieon due to the J_2 non-spherical gravity term for a central body.

Figure 13-17(f) presents a polar plot in which the radial axis is orbit inclination and the angle is RAAN. This plot shows the inclination and RAAN for Phobos' and Deimos' orbits and clearly demonstrates that at no time during the analysis interval are the orbits coplanar. Figure 13-18 further quantifies this by showing the angle between Phobos' and Deimo's orbit planes as a function of time; the angle between their planes is never zero during the analysis interval. In particular, the angle between the moons' orbit planes varies between 1.26^{o} and 3.76^{o} between the years 2030 and 2040, with a mean value of 2.67^{o}.

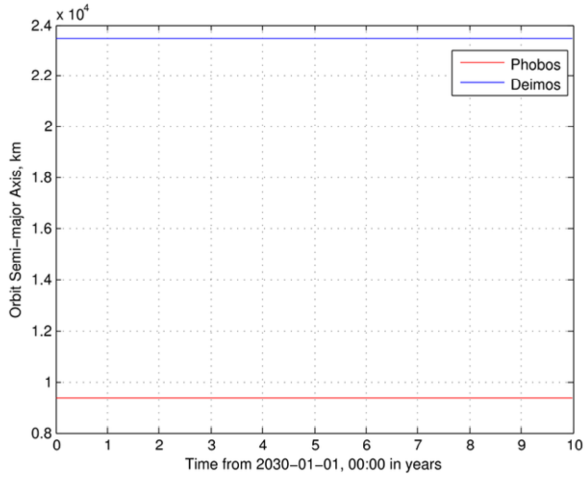
For reference, the minimum, mean, and maximum values of the classical Keplerian orbital elements are computed for each moon's orbit and provided for Phobos in Table 13-7 and Deimos in Table 13-8.

Table 13-7 Minimum, mean, and maximum values for Phobos' classical Keplerian orbital elements during the interval between 2030 and 2040.

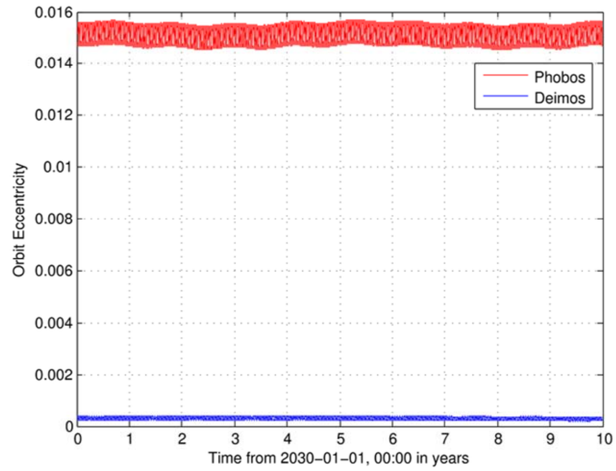
	a (km)	e	i	Ω	ω
Minimum	9377.75	0.0145	1.060°	0.002°	0.002°
Mean	9378.54	0.0151	1.075°	180.470°	181.075°
Maximum	9379.55	0.0157	1.091°	359.990°	359.999°

Table 13-8 Minimum, mean, and maximum values for Deimos' classical Keplerian orbital elements during the interval between 2030 and 2040.

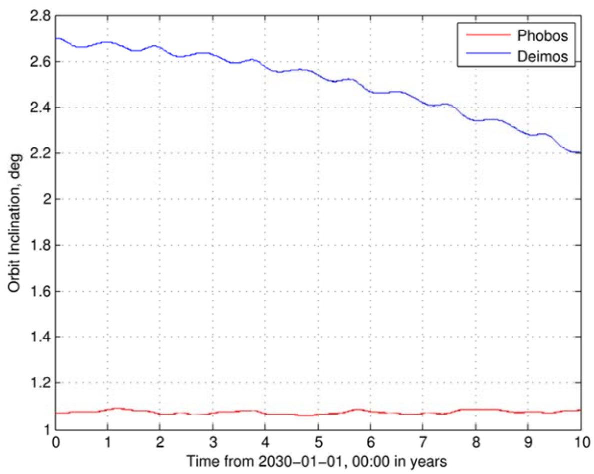
	a (km)	e	i	Ω	ω
Minimum	23457.92	0.00018	2.202°	171.970°	227.310°
Mean	23458.95	0.00030	2.506°	195.600°	289.333°
Maximum	23459.92	0.00041	2.697°	217.940°	357.580°



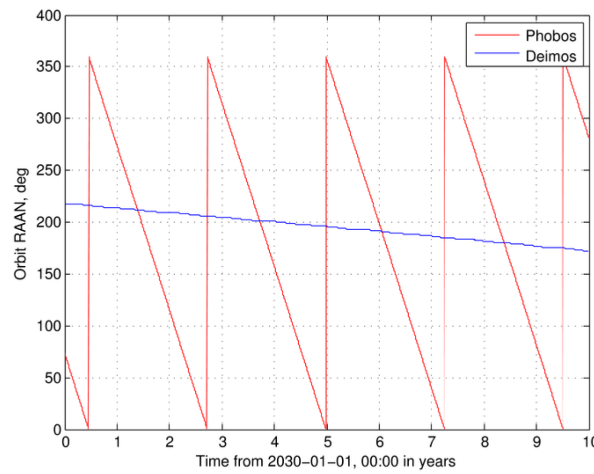
(a) Semi-major axis evolution



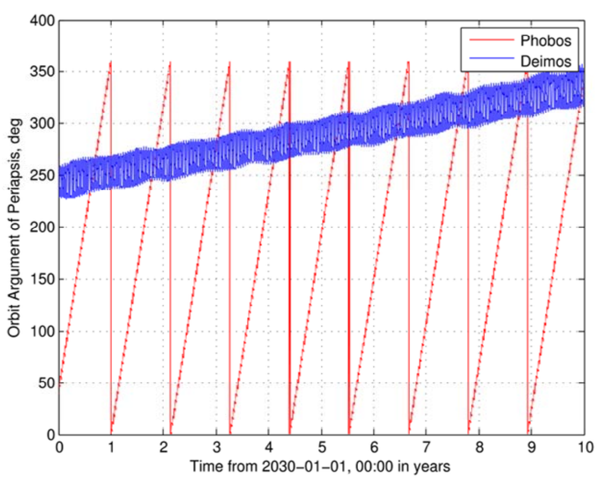
(b) Eccentricity evolution



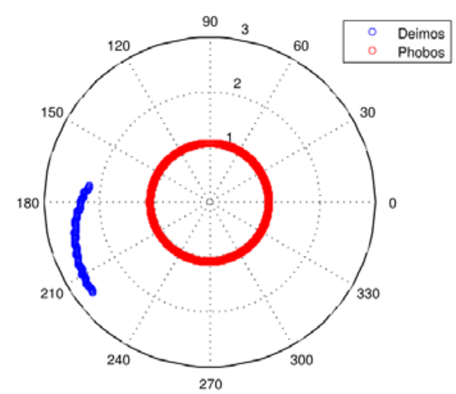
(c) Inclination evolution



(d) Right ascension of the ascending node (RAAN) evolution



(e) Argument of periapsis evolution



(f) Polar plot of RAAN and inclination evolution

Figure 13-17 Time histories of Phobos and Deimos orbital elements

One of the next topics addressed is optimal rendezvous trajectories between the moons. The synodic period between the moons was computed since it was expected that the optimal rendezvous trajectory opportunities to repeat approximately according to the synodic period. The stability of the semi-major axes of the moons' orbits makes their synodic period relatively stable at around 10.251 hours as shown in Figure 13-19.

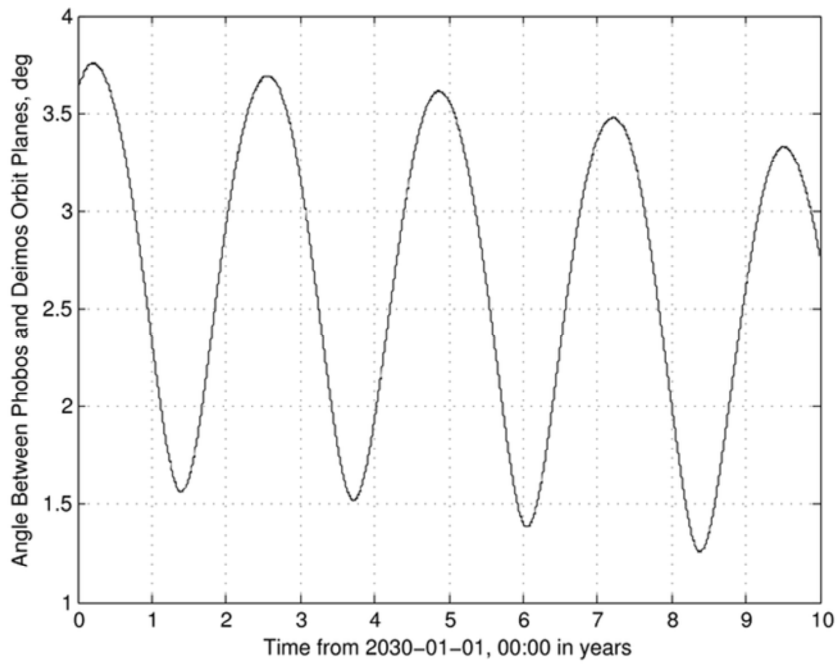


Figure 13-18 Angle between Phobos and Deimos orbit planes.

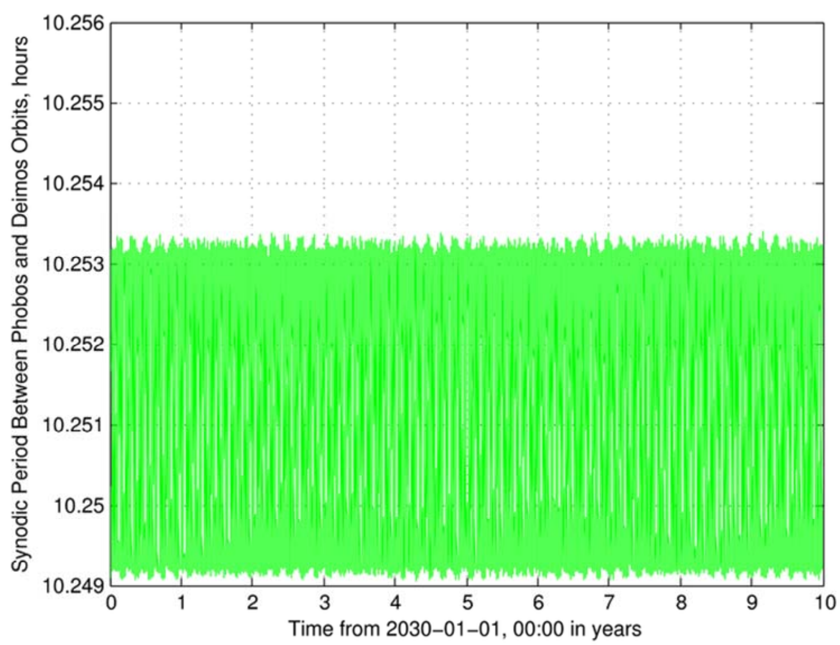


Figure 13-19 Evolution of the synodic period between Phobos and Deimos.

13.1.2.2. Rendezvous Trajectory Optimization for Phobos and Deimos

The precise ephemerides for Phobos' and Deimos' orbits were used to evaluate optimal (minimum Δv) two-impulse rendezvous trajectories between the moons. It is unlikely that the total rendezvous Δv can be reduced by using more complicated maneuvering schemes involving more than two maneuvers because the ratio of Phobos' and Deimos' orbit radii is approximately 2.5 and the angle between their orbit planes is less than 3.76^{o} throughout the analysis interval. The optimal two-impulse rendezvous solutions were identified using a trajectory grid search technique operating on the precise ephemerides of the moons. Small grids, each spanning several days of departure times, were scanned on the first day of each year between 2030 and 2040 using a Lambert solver algorithm to determine the initial and final Δv for each rendezvous trajectory. When computing these maneuvers it was assumed that each moon's gravity is weak enough to ignore (i.e., patched conics for arriving at or departing from the moons are not necessary).

To provide a basis for comparison the Hohmann transfer Δv between the moons was computed, including plane change optimally split between the initial and final maneuvers. These Hohmann transfer results were computed throughout the entire analysis interval using the precise semi-major axis values shown in Figure 13-17(a) as the radii for each moon's orbit (the orbits were treated as circular and ignore eccentricity) and using the precise angle between the moons' orbit planes shown in Figure 13-18.

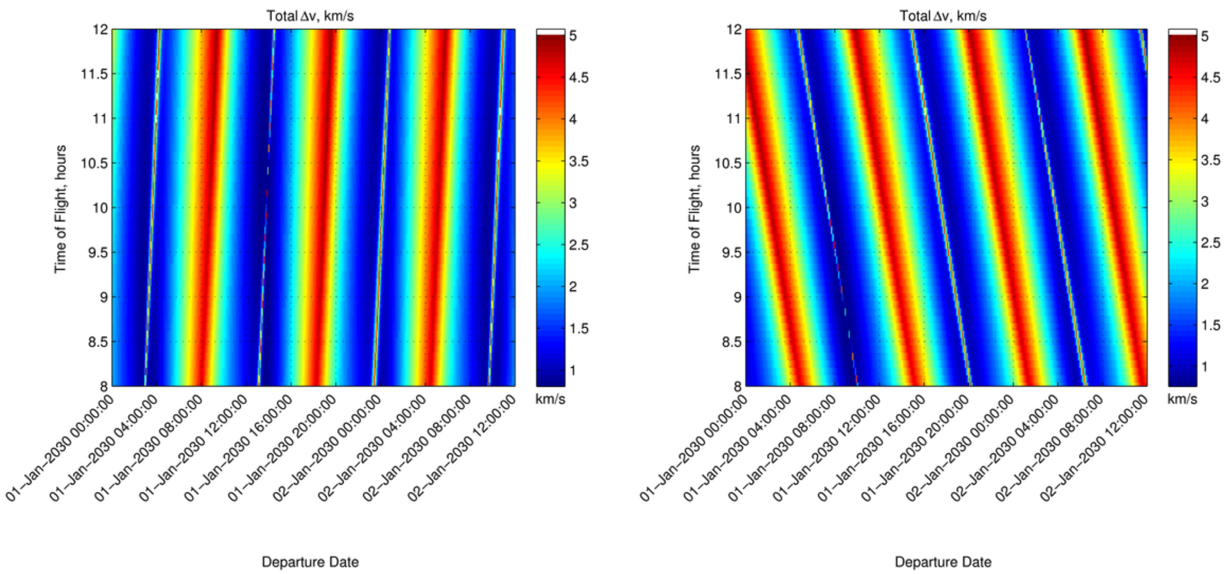
Figure 13-20(a) and 6(b) show the Pork Chop Contour (PCC) plots for the Phobos to Deimos and Deimos to Phobos trajectory scans, respectively. Note that the optimal rendezvous opportunities between the moons generally repeat according to the aforementioned synodic period between the moons (10.25 hours). Also note that the total Δv required for rendezvous increases steeply for non-optimal departure times.

Figure 13-21(a) and 7(b) show example optimal rendezvous trajectories from Phobos to Deimos and Deimos to Phobos, respectively, in a three-dimensional perspective view. Figure 13-22(a) and 8(b) show the projections of these trajectories onto Mars' equatorial plane. Note that while these optimal trajectories are Hohmann- like, they are clearly not exactly Hohmann transfers.

Figure 13-23(a) and 9(b) present the comparison between the precise optimal rendezvous Δv results from the grid scans and the approximate optimal rendezvous Δv results from the Hohmann transfer calculations. The change in the angle between the moons' orbit planes over time drives a mild variation over time in the Hohmann transfer Δv result, although the flight time yielded by the Hohmann calculations is always just less than 9 hours.

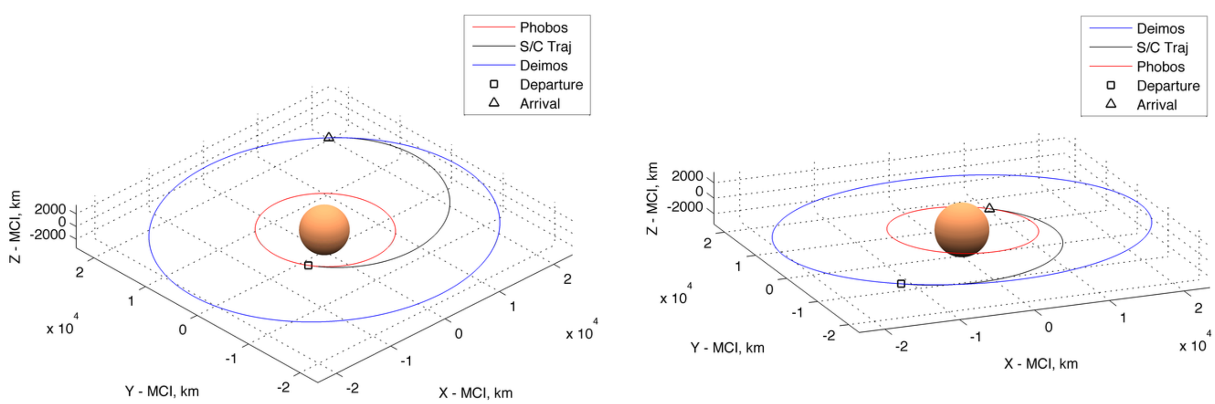
The optimal Δv solutions identified by the trajectory grid scans occasionally agree well with the approximate Hohmann transfer result, but this is generally not the case. Figure 13-23 (a) shows that the precise optimal Δv is generally slightly larger than the Hohmann result, up to 8% larger. Figure 13-23 (b) shows that corresponding precise flight times for the optimal rendezvous trajectories vary between about 10% shorter and 18% longer than the Hohmann transfer flight time. The periodic variations in the precise optimal rendezvous Δv and flight time results are chiefly driven by the change in the angle between the moons' orbit planes over time, as well as the fluctuation of Phobos' orbit eccentricity over time.

The overall maximum precise optimal total rendezvous Δv solution in Figure 13-23 (a) is 816 m/s with an associated flight time of 10.42 hours, and the overall minimum precise optimal total rendezvous Δv solution is 751 m/s with a flight time of 8.75 hours.



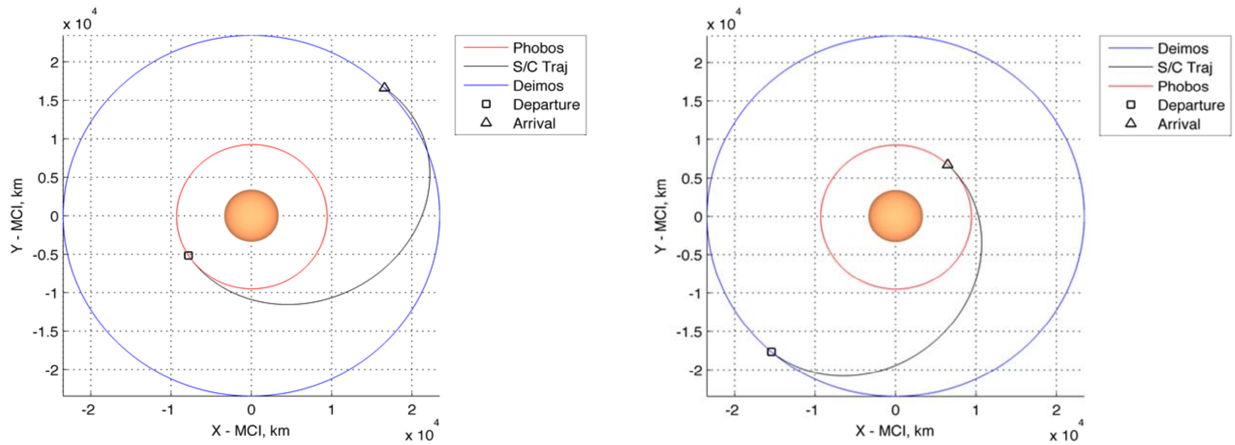
(a) Phobos to Deimos (b) Deimos to Phobos

Figure 13-20 Pork Chop Contour (PCC) plots for rendezvous trajectories between Phobos and Deimos using trajectory.



(a) Phobos to Deimos (b) Deimos to Phobos

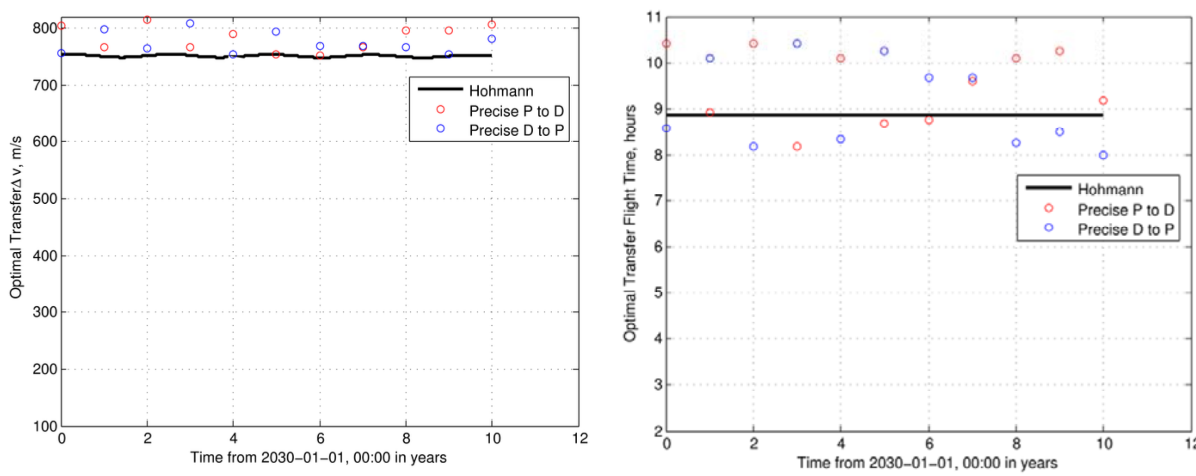
Figure 13-21 Example optimal two-impulse rendezvous trajectories between Phobos and Deimos, 3D view.



(a) Phobos to Deimos

(b) Deimos to Phobos

Figure 13-22 Example optimal two-impulse rendezvous trajectories between Phobos and Deimos, Mars equatorial plane.



(a) Phobos to Deimos

(b) Deimos to Phobos

Figure 13-23 Comparison of optimal rendezvous Δv and flight time between Phobos and Deimos for the precise Lambert

13.1.2.3. Terminal Rendezvous for Phobos and Deimos

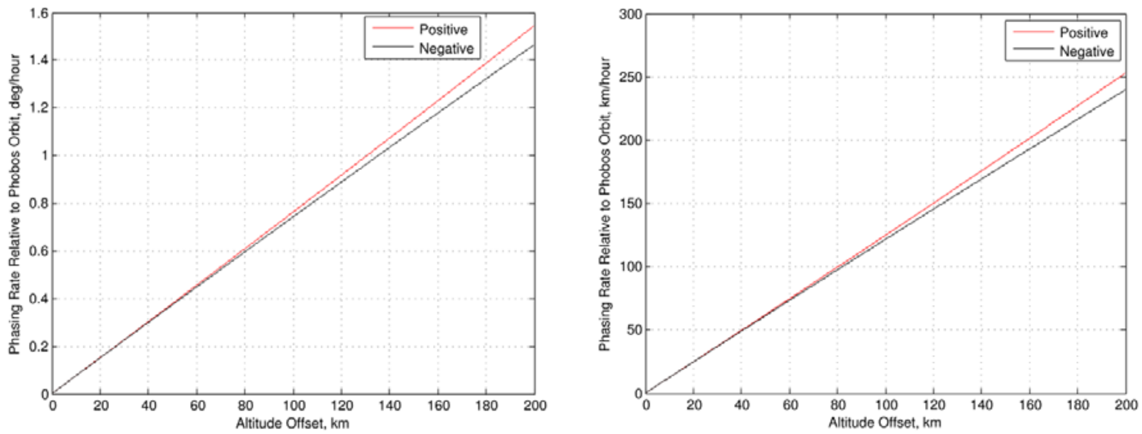
The optimal rendezvous trajectories discussed previously were calculated by targeting each moon directly, and the spacecraft velocity relative to each moon upon arrival is generally ~400 m/s. While those conditions are appropriate for grid scans to design optimal orbital rendezvous trajectories, in practice the spacecraft will require a controlled flight path relative to each moon with gradual approach speeds that are conducive to safety.

It was then assumed that the orbital rendezvous trajectories shown previously were actually targeting a co-elliptic orbit[3] with respect to the destination moon's orbit, and that the location targeted on the co-elliptic orbit would be slightly offset in the in-track direction relative to the moon. This scenario would allow the spacecraft to naturally drift towards the destination moon after achieving co-elliptic orbit, and the drift rate relative to the destination moon

[3] A co-elliptic orbit is coplanar with the target orbit and has the same eccentricity but a slightly different semi-major axis.

would be regulated by occasional pairs of small maneuvers to adjust the altitude of the spacecraft's orbit. This sequence of drift and altitude adjustment segments beginning at the offset location on the co-elliptic orbit is referred to as terminal rendezvous.

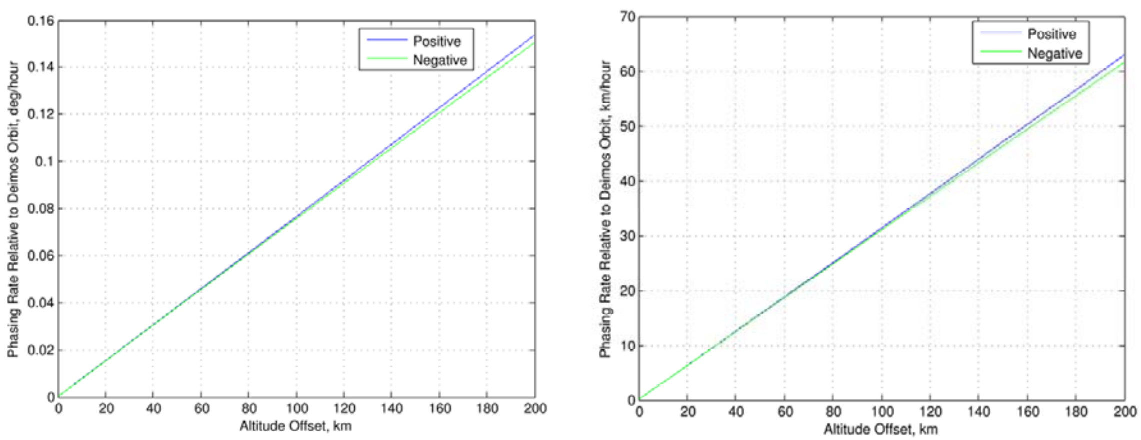
This terminal rendezvous strategy would provide safety and leverages prior human space flight rendezvous operations experience (e.g., Space Shuttle rendezvous with the International Space Station (ISS)) at the expense of increasing the amount of time required to complete the rendezvous. The available drift rates on a co-elliptic orbit relative to Phobos as a function of altitude difference are presented in Figure 13-24(a) and 10(b) in units of degrees per hour and kilometers per hour, respectively. Figure 13-25(a) and 11(b) provide these data for Deimos. Note that Phobos' lower orbit admits to higher relative phasing rates than does Deimos' higher orbit for equivalent altitude offsets. This feature of the relative motion dynamics means that terminal rendezvous with Deimos will generally require more time than terminal rendezvous with Phobos unless additional Δv is employed to achieve more drastic co-elliptic orbit altitude offsets relative to Deimos' orbit.



(a) Phasing rates relative to Phobos in degrees per hour.

(b) Phasing rates relative to Phobos in km per hour.

Figure 13-24 Phasing rates relative to Deimos as a function of altitude offset from Phobos' orbit.



(a) Phasing rates relative to Deimos in degrees per hour.

(b) Phasing rates relative to Deimos in km per hour.

Figure 13-25 Phasing rates relative to Deimos as a function of altitude offset from Phobos' orbit.

Terminal trajectory sequences for rendezvous with each moon were then considered. These trajectory sequences were constructed using the Hill's/Clohessy-Wiltshire (HCW) equations of relative orbital motion, assuming each moons' orbit to be circular. This is a reasonable assumption in light of the relatively small eccentricity of each moons' orbit and permits rapid trajectory design using the HCW equations.

Terminal rendezvous with Phobos would begin at an in-track distance of 1000 km behind Phobos with an altitude 75 km below Phobos' orbit. Four drifting segments were defined, with altitude raises between them, that would bring the spacecraft to a point 15 km directly below Phobos 27.42 hours after the initiation of terminal rendezvous for a total terminal rendezvous v cost of 6.84 m/s. The drift and altitude raise segments, along with the associated individual flight times and maneuver magnitudes, are presented in Table 13-9.

Figure 13-26 shows the Phobos terminal rendezvous trajectory sequence in the radial, in-track plane of the radial, in-track, cross-track (RIC) frame whose origin is at Phobos' center of mass. The trajectory plot in Figure 13-26 uses equal plot axis scaling to show the actual appearance of the trajectory, but this makes it difficult to discern the geometrical structure of the trajectory. Figure 13-27(a) offers an alternative view of the Phobos terminal rendezvous trajectory with unequal plot axis scaling that distorts the view of the trajectory (and Phobos itself) but aids in clarifying the relative motion geometry. Figure 13-27(b) uses equal plot axis scaling to avoid distortions and shows the end of the terminal rendezvous trajectory arriving directly below Phobos at a distance of 15 km. At this point the spacecraft may perform additional maneuvers to begin proximity operations about Phobos.

Table 13-9 Example terminal rendezvous sequence for Phobos.

Segment Type	Altitude Change (km)	Flight Time (hours)	ΔV (m/s)
Drift	0	2.00	N/A
Altitude Raise	-75 to -30	3.83	2.56
Drift	0	4.50	2.56
Altitude Raise	-30 to -20	3.83	0.57
Drift	0	4.50	0.57
Altitude Raise	-20 to -15	3.83	0.28
Drift	0	4.93	0.28
TOTALS		27.42	6.84

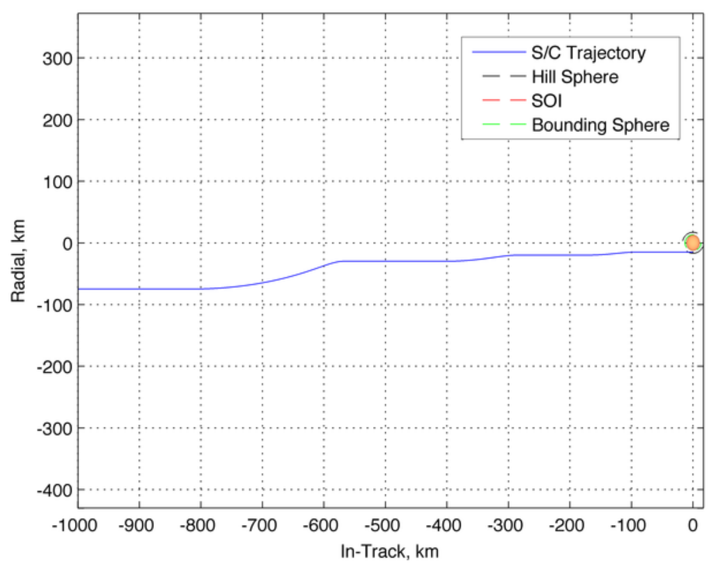
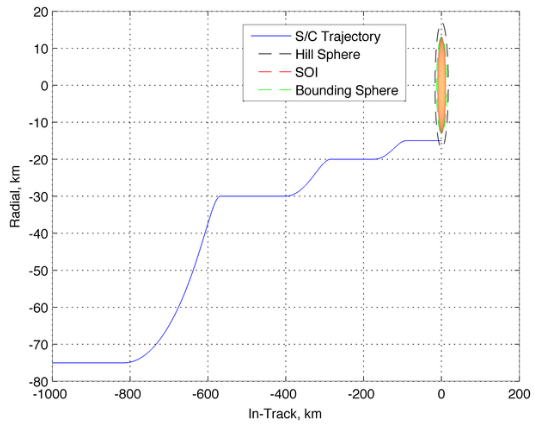
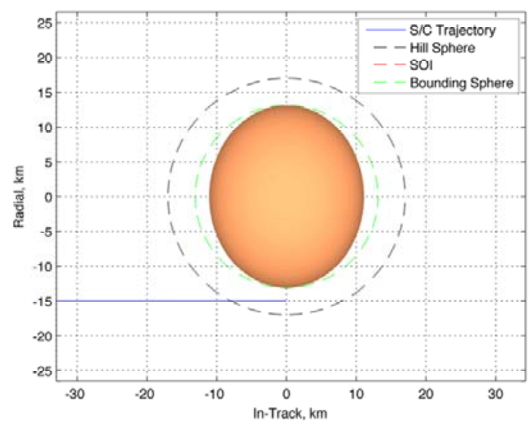


Figure 13-26 Example terminal rendezvous trajectory for Phobos.



(a) Example terminal rendezvous trajectory for Phobos, unequal plot axis scaling..



(b) Detail view of the end of the Phobos terminal rendezvous trajectory.

Figure 13-27 Example Phobos terminal rendezvous trajectory shown with unequal plot scaling, along with a detailed view of the end of the trajectory.

Terminal rendezvous with Deimos would begin at an in-track distance of 1000 km behind Deimos and with an altitude 100 km below Deimos' orbit. This altitude offset is slightly larger than for the Phobos case presented previously because of the generally slower phasing rates relative to Deimos at a given altitude offset. Four drifting segments were defined, with altitude raises between them, that would bring the spacecraft to a point 12 km directly below Deimos 79.55 hours after the initiation of terminal rendezvous for a total terminal rendezvous Δv cost of 2.53 m/s. The drift and altitude raise segments, along with the associated individual flight times and maneuver magnitudes, are presented in Table 13-10.

Figure 13-28 shows the Deimos terminal rendezvous trajectory sequence in the radial, in-track plane of the radial, in-track, cross-track (RIC) frame whose origin is at Deimos' center of mass. The trajectory plot in Figure 13-28 uses equal plot axis scaling to show the actual appearance of the trajectory, but this makes it difficult to discern the geometrical structure of the trajectory. Figure 13-29(a) offers an alternative view of the Deimos terminal rendezvous trajectory with unequal plot axis scaling that distorts the view of the trajectory (and Deimos itself) but aids in clarifying the relative motion geometry. Figure 13-29(b) uses equal plot axis scaling to avoid distortions and shows the end of the terminal rendezvous trajectory arriving directly below Deimos at a distance of 12 km. At this point the spacecraft may perform additional maneuvers to begin proximity operations about Deimos.

Table 13-10 Example terminal rendezvous sequence for Deimos.

Segment Type	Altitude Change (km)	Flight Time (hours)	ΔV (m/s)
Drift	0	3.00	N/A
Altitude Raise	-100 to -65	15.15	0.50
Drift	0	5.00	0.50
Altitude Raise	-65 to -25	15.15	0.58
Drift	0	5.00	0.58
Altitude Raise	-25 to -12	15.15	0.19
Drift	0	21.09	0.19
TOTALS		79.55	2.53

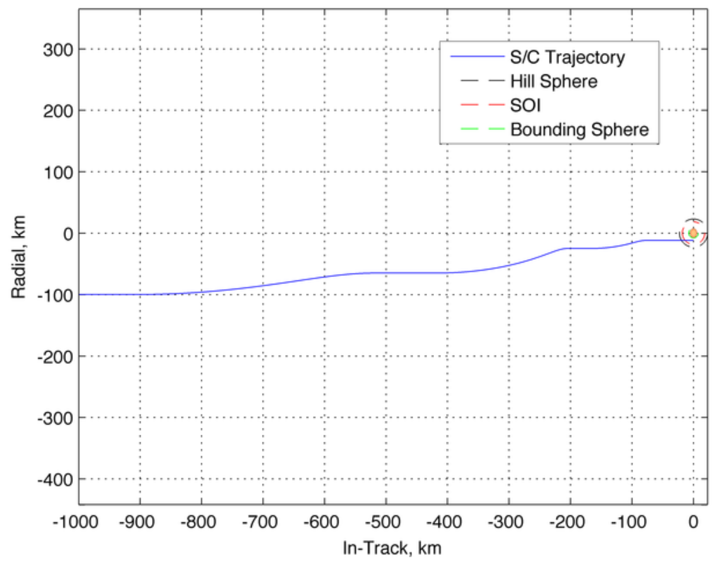
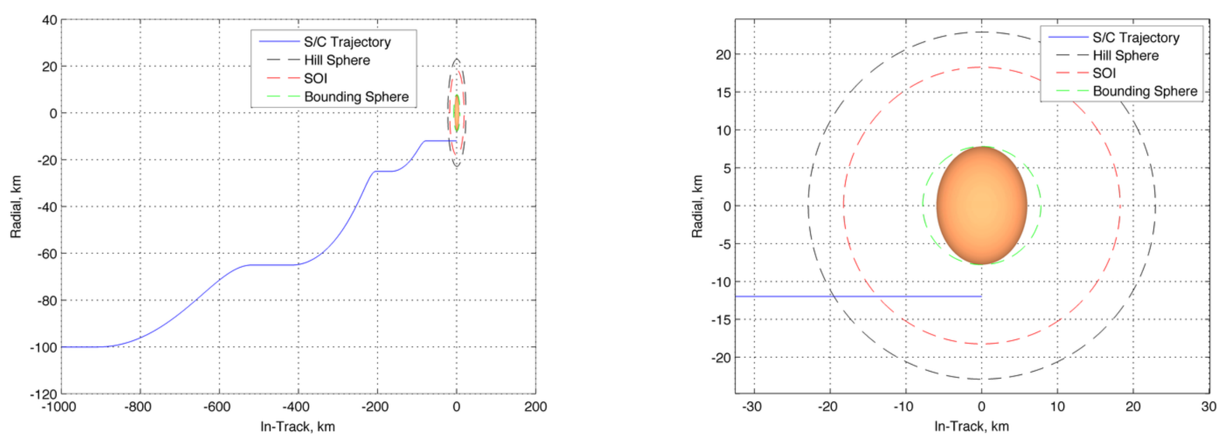


Figure 13-28 Example terminal rendezvous trajectory for Deimos.



(a) Example terminal rendezvous trajectory for Phobos, unequal plot axis scaling..

(b) Detail view of the end of the Phobos terminal rendezvous trajectory.

Figure 13-29 Example Deimos terminal rendezvous trajectory shown with unequal plot scaling, along with a detailed view of the end of the trajectory.

The Δv required for terminal rendezvous would be quite manageable for both Phobos and Deimos, though the terminal rendezvous sequences would be somewhat time-consuming (a little more than a day for Phobos and a little over 3 days for Deimos). As noted previously, the time required to phase to Deimos would be longer than that for Phobos unless larger altitude offsets are employed. However, the use of larger altitude offsets would require additional Δv and will also alter the relative geometry during major portions of the approach. The impact of alternative relative motion geometries during approach on sensors and relative navigation filters would need to be examined. Hohmann transfers are used for all of the altitude raises in order to minimize v requirements, but shorter (non-Hohmann) transfers may be fl own for altitude raises to reduce overall flight time for terminal rendezvous at the expense of increasing total terminal rendezvous Δv. That being said, note that the time between altitude raise maneuvers would be already relatively short for many segments, often on the order of only several hours, and this may cause issues with relative navigation filter convergence between maneuvers. Other factors not considered here, such as the impact of lighting condition on relative navigation sensor performance, would influence actual terminal rendezvous trajectory design in practice.

13.1.2.4. Orbital Operations About Phobos and Deimos

Upon arriving at either moon at the end of terminal rendezvous, the spacecraft would begin proximity operations about the moon. During this phase the primary crew vehicle may collect sensor data and deploy crew in excursion vehicles or individual maneuvering suits to interact with sites on the moon's surface. A strategy would therefore be required for maintaining some sort of proximity operations posture relative to the moon while the aforementioned activities would be taking place. This analysis started by assessing each moon's ability to support captured orbits. Viable proximity operations strategies may involve captured orbits, forced motion (e.g., station-keeping or travel between waypoints), or more some combination thereof.

13.1.2.4.1. Sphere of Influence / Hill Sphere Radii for Phobos and Deimos

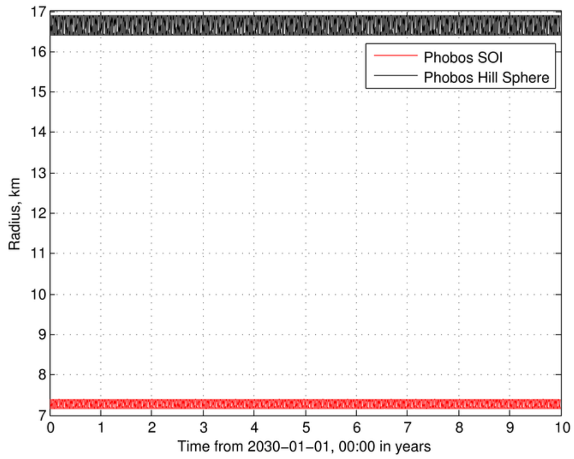
The restricted three-body system consisting of Mars, Phobos or Deimos, and the spacecraft was first considered, assessing each moon's Sphere of Influence (SOI) and Hill sphere. Those volumes each provide an estimate for the region of space surrounding the moon in which stable captured orbits may be achieved. When perturbations beyond point mass gravity for the moon and Mars are included, the spatial volumes around each moon within which stable motion would be possible are found to not be spherical, in general; however, those effects are not considered herein.

In a system consisting of a large celestial body and a smaller one, the Hill sphere is the approximate region in which the smaller body's gravity is more dominant than that of the larger body; thus, the smaller body may theoretically have captured satellites within the Hill sphere radius. The Hill sphere radius is also the distance of the L_1/L_2 Lagrangian points from the smaller celestial body. Note that the Hill sphere may also be referred to as the Sphere of Activity, Activity Sphere, or Roche Sphere (not to be confused with the Roche Limit). The SOI is simply a different (and usually more conservative) way to describe the region in which a smaller body is the primary gravitational influence on satellites. The SOI radius is defined mathematically as the distance from the smaller body at which it becomes appropriate to treat the smaller body's gravity as a perturbation and treat the larger body as the primary attractor (within the SOI radius, the situation is reversed).

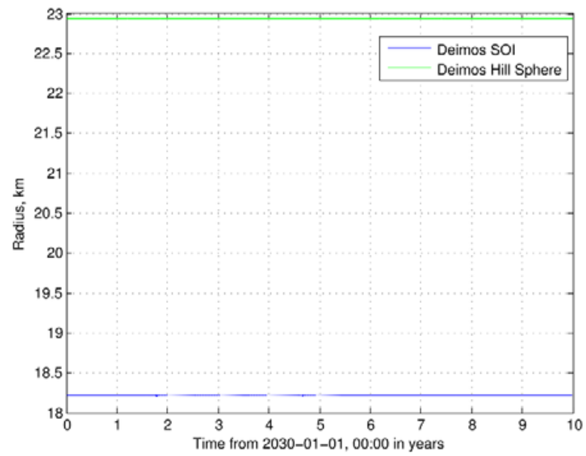
The SOI and Hill sphere radii throughout the analysis interval are shown in Figure 13-30(a) for Phobos and Figure 13-30(b) for Deimos. Note that Deimos' SOI and Hill sphere radii are more stable than those of Phobos because of Deimos' lower orbit eccentricity (Deimos' distance to Mars fluctuates less over time). Conversely, the observed variation in Phobos' SOI and Hill sphere radii are due to its orbit eccentricity. As indicated previously, the SOI radii for the moons are smaller than their Hill sphere radii and therefore provide a more conservative assessment of the space relative to each moon within which stable captured orbits may be possible.

Comparing the physical dimensions of the moons, provided in Figure 13-15 to their SOI and Hill sphere radii shows that Phobos' SOI radius is only 55% of its physical bounding radius and its Hill sphere radius is only larger than its SOI radius by a factor of 2.33. This is a strong indication that achieving stable captured orbits around Phobos may be rather difficult, if not impossible. By contrast, Deimos' SOI radius is 2.34 times larger than its physical bounding radius and its Hill sphere radius is 1.25 times larger than its SOI radius. This indicates that stable captured orbits about Deimos are likely to be possible. Note that the SOI and Hill sphere outlines for Phobos and Deimos are depicted in Figure 13-27(b) and Figure 13-29(b), respectively.

These observations were then investigated further by attempting to design stable captured orbits around each moon using an elliptical restricted three-body dynamics model in which both Mars and the moon are treated as point masses for the purpose of gravity field modeling. Each moon's orbit around Mars was modeled using the mean classical Keplerian orbital elements presented in Table 13-7 and Table 13-8. Gravitational parameters for each moon are computed by scaling their masses from Figure 13-15 by the Universal Gravitational Constant, G = 6:67259 x 10^{-20} $km^3s^{-2}kg^{-1}$. The gravitational parameter value used for Mars is 4:2828374747780377 x 10^4 km^3s^{-2}.



(a) Phobos SOI and Hill Sphere radii.



(b) Deimos SOI and Hill Sphere radii.

Figure 13-30 Sphere of Influence and Hill Sphere radii of Phobos and Deimos.

13.1.2.4.2. Captured Orbits About Deimos

Experimentation with prograde motion about Deimos resulted in short-lived orbits that would quickly destabilize and either escape Deimos or collide with its surface. Retrograde motion about Deimos proves to be much more stable, and Figure 13-31 shows an example stable retrograde orbit about Deimos in a Deimos-centered inertial frame. The initial orbit radius is 9 km, the initial orbit period is 4.3 hours (0.18 days), and the initial orbit velocity is 3.65 m/s. Note that the magnitude of the orbit velocity is of the same order as the relative motion velocity during the terminal rendezvous phase and thus transitioning to a stable captured orbit from terminal rendezvous would generally require only a few m/s of v. The stable retrograde orbital motion is shown for a time-span of 20 days in Figure 13-31 but may remain stable for longer than that (this was not investigated).

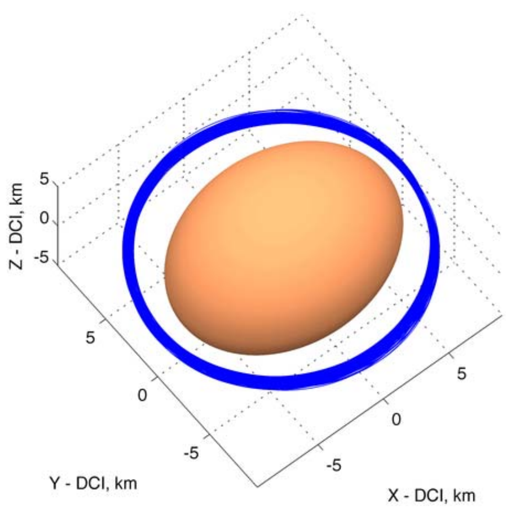


Figure 13-31 Example Deimos-captured orbit (retrograde) propagated for 20 days.

13.1.2.4.3. Captured Orbits About Phobos

As was the case for Deimos, prograde motion about Phobos proved to be unstable in the simulation. However, making the motion retrograde did not significantly improve matters for Phobos; an example of a stable captured orbit about Phobos in the simulation was not identified. All of the initial conditions attempted would result in either immediate escape or collision with Phobos' surface. Figure 13-32 shows an example of an escape trajectory in a

Phobos-centered inertial frame. The initial orbit radius is 15 km, the initial orbit period is 3.8 hours (0.16 days), and the initial orbit velocity is 6.93 m/s. Figure 13-32 only shows 2.4 hours of the simulated motion.

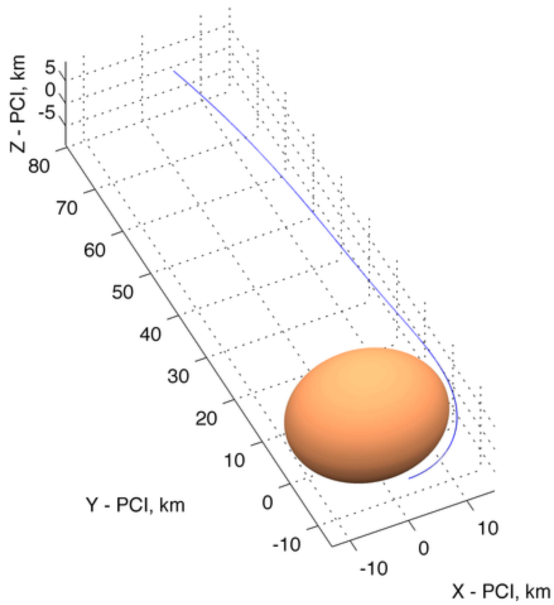


Figure 13-32 Example Phobos-captured orbit attempt propagated for 2.4 hours.

Achieving captured retrograde orbits around Deimos appears feasible, and such orbits may be good locations from which to stage crew excursions to/from Deimos although additional analysis is required. Captured orbits around Phobos would be more problematic, however. Phobos is much closer to the primary body (Mars) and as such its SOI radius is well inside its physical bounding sphere, making stable captured orbits difficult to identify. Pseudo-captured orbits about Phobos could probably be achieved with frequent maintenance maneuvers, and such maneuvers would almost certainly be more costly than for Deimos-captured orbit maintenance. The level of control effort and frequency required would likely increase with altitude above Phobos and may simply amount to forced motion. In this case, station-keeping at a standoff location may be more economical. In any case, the Δv to transition from co-elliptic drift to captured orbits should be small in general (on the order of several m/s). Finally, it is worth noting that further analysis should be performed in which non-spherical gravity field models are used for the moons and Mars, in combination with other natural perturbations (such as solar radiation pressure). In those models it may be possible to identify sufficiently stable motion patterns relative to the moons (even Phobos) that would be suitable for operations even if those motion patterns do not constitute orbits in the traditional sense; periodicity and stability may be more important than circumnavigation of the moon.

13.1.2.5. Maneuver Sequence for Exploring Phobos and Deimos

The previous results were then integrated with hyperbolic arrival and departure conditions at Mars to synthesize a complete maneuvering sequence for exploring Phobos and Deimos that includes arrival in a highly elliptical Mars orbit from an incoming hyperbolic approach trajectory from interplanetary space, maneuvers to reorient the highly elliptical Mars orbit for subsequent departure, exploration of both Phobos and Deimos, and return to the highly elliptical Mars orbit for departure.

The maneuver calculations utilize the results of the preceding sections in combination with a subset of broad round-trip interplanetary trajectory scans involving various stay times at Mars. 9267 round-trip trajectories were initially generated, from which 142 were selected that require less total mission mass to utilize for the results presented herein. The round-trip trajectories considered include opposition class trajectories with stay times at Mars between 20 and 100 days, and conjunction class trajectories with stay times at Mars of 480 - 580 days. Several types of opposition class trajectories were considered: Earth-Mars-Earth (EME), Earth-Mars-Venus-Earth (EMVE), and Earth-Venus-Mars-Earth (EVME). The EME trajectories fly directly to Mars and then directly back to Earth. The

EMVE trajectories include a Venus gravity assist on the way back to Earth from Mars, and the EVME trajectories include a Venus gravity assist on the way to Mars from Earth. The conjunction class round-trip trajectories tend to feature lower overall total mission Δv compared to the opposition class round-trip trajectories, while the opposition class trajectories tend to feature lower overall mission durations. Including Venus gravity assists in the opposition class trajectory sequences can help reduce the total mission Δv in some cases.

The highly elliptical orbit into which the spacecraft would initially capture at Mars arrival, and from which the spacecraft would depart Mars after Phobos/Deimos exploration, has a periarieon altitude of 250 km and an apoarieon altitude of 33813 km, which corresponds to a semi-major axis of 20421.4 km, an eccentricity of 0.82176, and a period of 1.025483 days (approximately 1 sol). Additionally the maneuver calculations assume the aforementioned Mars gravitational parameter of 4:2828374747780377 x10^4 km^{3}s^{-2} and a mean Mars radius of 3389.9 km. Furthermore, the mean orbits of Phobos and Deimos presently previously were assumed

13.1.2.5.1. Maneuver Sequence Overview

The first step in the maneuver sequence would be to enter the highly elliptical capture orbit at periarieon on the incoming hyperbola at Mars. Next, the spacecraft would coast to the apoarieon of the capture orbit where it would perform a plane change maneuver to reorient the orbit plane as to be properly aligned for Mars departure. It assumed that the crew would leave the majority of their spacecraft stack in the departure orbit and utilize a smaller craft to explore Phobos and Deimos so as to minimize the amount of mass brought deeper into Mars' gravity well and thereby reduce overall propellant mass requirements for the mission. For the purposes of this overview it was also assumed that Deimos would be visited first (though the Phobos-first case was analyzed as well). The smaller spacecraft would remain on what would be refered to as the departure orbit for one period, after which it would perform a maneuver at apoarieon to match Deimos' orbit plane and simultaneously raise periarieon to match the radius of an orbit that is co-elliptic with Deimos' orbit. The spacecraft would then coast to the co-elliptic orbit radius at the periarieon of the transfer ellipse and matche the orbital velocity of the co-elliptic orbit. Next, the spacecraft would perform terminal rendezvous with Deimos as per the terminal rendezvous maneuver sequence presented previously. After exploring Deimos, it was assume a wait time at Deimos of at least one Phobos/Deimos synodic period, after which the spacecraft would fly the previously presented optimal rendezvous trajectory to an orbit that is co-elliptic with Phobos' orbit. Terminal rendezvous with Phobos would then performed, followed by exploration of Phobos for some period of time. After this, the spacecraft would perform a maneuver to raise apoarieon to match that of the departure orbit (timing this maneuver so as to intercept the spacecraft stack on the departure orbit at apoarieon). The spacecraft would then coast to the departure orbit apoarieon on the transfer ellipse, perform a maneuver to simultaneously match the departure orbit plane and periarieon radius, and then reconnect with the spacecraft stack at apoarieon. The spacecraft stack would then coast to the departure orbit periarieon and perform a maneuver at periarieon to inject into the outbound hyperbola to depart Mars.

13.1.2.5.2. Example Maneuver Sequence Results

The maneuver magnitudes and flight times associated with each of the segments described in the maneuver sequence overview were then computed assuming an incoming and outgoing V_∞ at Mars of 3 km/s, an asymptotic declination of 10^o at Mars arrival, and an asymptotic declination of 30^o at Mars departure. The maneuver sequence calculations for the Deimos-first case are presented in Table 13-11.

The results in Table 13-11 indicate that the Δv budget for the maneuvers performed at Mars[4] in this example is about 4.8 km/s and the total associated flight time for maneuvers and transfers is about 9 days. Thus the stay time at Mars must be long enough to accommodate the time spent exploring at each moon (not listed here because it is unknown) as well the approximately 9 days required for maneuvers and transfers to move about between the moons and the highly elliptical Mars capture/departure orbit.

[4] It is important to recognize that the total Δv at Mars does not include the other major maneuvers for the round-trip mission, which are the maneuver to depart Earth for Mars, any deep space maneuvers that may be performed on the way to Mars or on the way back home to Earth, and any maneuvers that may be required at Earth return to control the direct atmospheric re-entry speed of the crew vehicle or capture it into an Earth orbit.

Table 13-11 Example maneuver sequence in which Deimos is visited first, then Phobos.

Segment Type	DV (m/s)	Flight Time (hours)
Enter Capture Orbit at Periarieon of Incoming Hyperbola	1073.9	
Coast to Capture Orbit Apoarieon		12.31
Match Departure Orbit Plane at Apoarieon	157.3	
Coast to Departure Orbit Apoarieon		24.61
Match Deimos' Orbit Plane and Raise Periarieon to Deimos' Orbit Radius	580.7	
Coast to Deimos' Orbit Radius at Periarieon of Transfer Ellipse		22.75
Match Deimos' Orbital Velocity at Periarieon of Transfer Ellipse	145.3	
Perform Terminal Rendezvous with Deimos	2.5	79.55
Wait for Synodic Period to Elapse to Fly Optimal Transfer to Phobos		10.25
Fly Optimal Transfer to Phobos' Orbit	816	10.42
Perform Terminal Rendezvous with Phobos	6.8	27.42
Raise Apoparieon to Match Departure Orbit Apoarieon	563.8	
Coast to Departure Orbit Apoarieon on Transfer Ellipse		14.99
Match Departure Orbit Plane and Periarieon Radius	359	
Coast to Departure Orbit Periarieon		12.31
Exit Departure Orbit on Outbound Hyperbola	1073.9	
TOTALS	4779.2	214.13

Table 13-12 presents the maneuver sequence results for the case in which Phobos is visited first. It turns out that the total at-Mars Δv and associated flight time for maneuvers and transfers are the same as for the Deimos-first case (4.8 km/s and 9 days) although the distribution of Δv and flight time amongst the individual segments is naturally a bit different.

Note that the Δv to capture into Mars orbit from the incoming hyperbola and to depart Mars orbit by injecting into the outgoing hyperbola account for about 45% of the 4.8 km/s at Mars in the examples. For the results in Table 13-11 and Table 13-12 it was assumed a particular incoming/outgoing V_∞ at Mars for the sake of creating example results, but in practice the incoming/outgoing V_∞ can vary significantly, and this will naturally affect the total Δv at Mars. If the Mars capture/departure Δv is excluded, the total Phobos/Deimos exploration Δv from Table 13-11 and Table 13-12 is about 2.6 km/s. The other factors which will cause this Δv to vary are the incoming/outgoing asymptotic declinations.

Table 13-12 Example maneuver sequence in which Phobos is visited first, then Deimos.

Segment Type	DV (m/s)	Flight Time (hours)
Enter Capture Orbit at Periarieon of Incoming Hyperbola	1073.9	
Coast to Capture Orbit Apoarieon		12.31
Match Departure Orbit Plane at Apoarieon	157.3	
Coast to Departure Orbit Apoarieon		24.61
Match Deimos' Orbit Plane and Raise Periarieon to Deimos' Orbit Radius	359.0	
Coast to Deimos' Orbit Radius at Periarieon of Transfer Ellipse		14.99
Match Deimos' Orbital Velocity at Periarieon of Transfer Ellipse	563.8	
Perform Terminal Rendezvous with Deimos	26.8	27.42
Wait for Synodic Period to Elapse to Fly Optimal Transfer to Phobos		10.25
Fly Optimal Transfer to Phobos' Orbit	816.0	10.42
Perform Terminal Rendezvous with Phobos	2.5	79.55
Raise Apoparieon to Match Departure Orbit Apoarieon	145.3	
Coast to Departure Orbit Apoarieon on Transfer Ellipse		22.28
Match Departure Orbit Plane and Periarieon Radius	580.7	
Coast to Departure Orbit Periarieon		12.31
Exit Departure Orbit on Outbound Hyperbola	1073.9	
TOTALS	4779.2	214.13

13.1.2.5.3. Trajectory Scan Results

The maneuver sequence calculations described previously were then applied to the aforementioned trajectory scans for round-trip Mars mission trajectories. The total Δv at Mars across the set of round-trip trajectories is presented in Figure 13-33(a). Recall that a relatively low total Δv at Mars does not necessarily mean that the total mission Δv is also relatively low. That being said, the EME opposition class trajectories tend to have somewhat lower total Δv at Mars than do the other types of trajectories in the scans. Figure 13-33(b) shows the stay times at Mars for the set of trajectories.

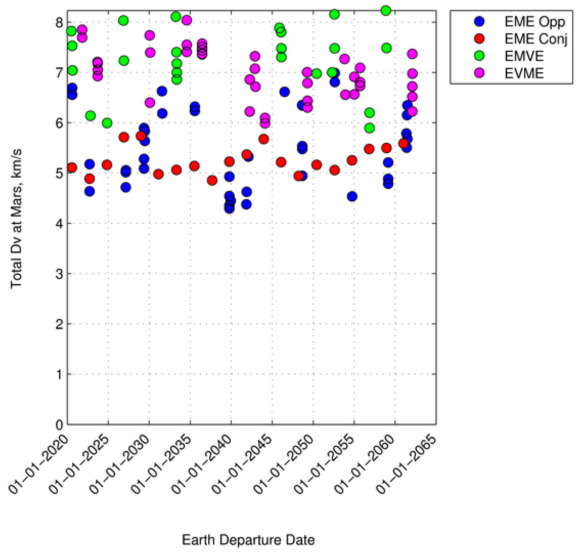
Figure 13-33(c) presents the relationship between the total mission duration and the total Δv at Mars. Note that while the EME opposition class trajectories have lower total Δv at Mars, they also have significantly longer total mission durations than the EMVE and EVME opposition class trajectories.

Figure 13-33(d) shows the differences between Mars arrival and departure asymptotic declinations for the set of trajectories. These differences are rather scattered, being small or near zero in some cases while being in excess of 25° to 30° in other cases. An arrival/departure declination difference of 20° were selected for the example calculations, shown previously in Table 13-11 and Table 13-12, because that seems to be a reasonably representative value.

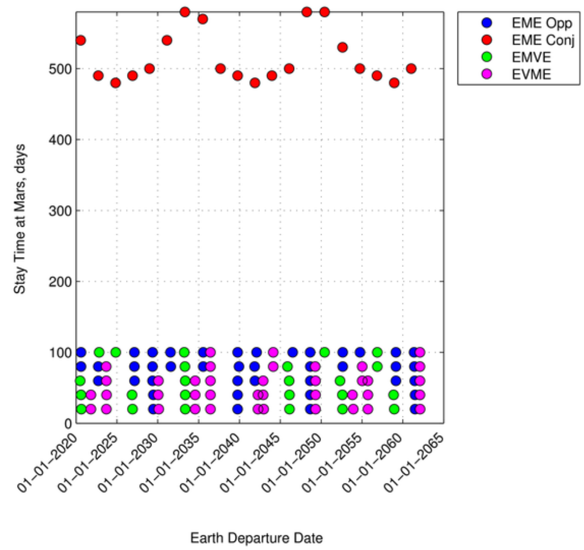
13.1.2.5.4. Effects of Arrival and Departure Asymptotic Declinations

In the example results shown previously, the total Δv at Mars for exploring Phobos and Deimos (not counting the Mars arrival and departure Δv, which depends solely on incoming/outgoing V∞) is 2.6 km/s, assuming a 10° asymptotic declination at Mars arrival and a 30° asymptotic declination at Mars departure (yielding a declination difference of 20°). The effects of other combinations of arrival and departure asymptotic declinations on the total Δv at Mars for exploring Phobos and Deimos were then investigated (again, independent of incoming/outgoing V∞ at Mars).

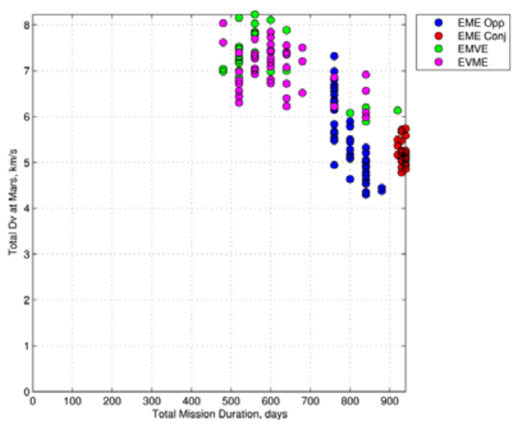
A parametric scan in which both arrival and departure asymptotic declination are systematically varied between 0° and 80° was then performed. The total Δv at Mars for exploring Phobos and Deimos for each combination of arrival and departure asymptotic declination was computed. The results of this parametric scan are presented in Figure 13-34. One interesting trend is that higher arrival declinations have less impact on the Δv when the departure declinations are lower.



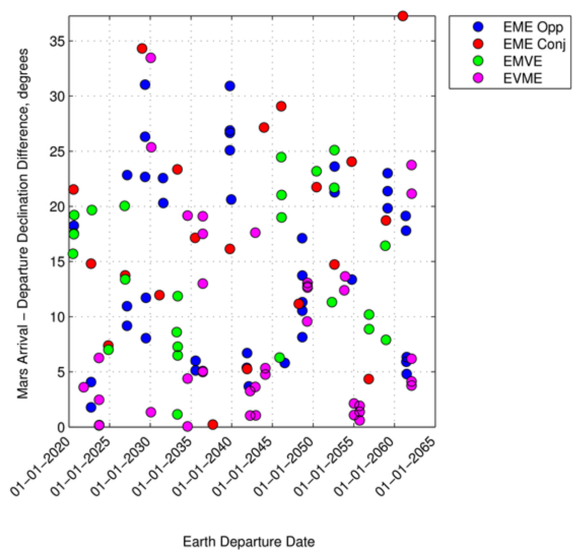
(a) Total ΔV at Mars.



(b) Stay time at Mars.



(c) Total round-trip mission duration versus total ΔV at Mars.



(d) Difference between Mars arrival and departure asymptotic declinations.

Figure 13-33 Trajectory scan results for total ΔV at Mars, stay time at Mars, mission duration, and differences between asymptotic declinations for Mars arrival and departure.

The total Δv at Mars between capture and departure shown in Figure 13-34 ranges between 2.25 km/s and 3.81 km/s. However, recall from Figure 13-33(d) that the largest arrival/departure asymptotic declination difference seen in our round-trip trajectory scans is approximately 38^{o} and so it is not expected to be in the upper portion of the Δv range shown in Figure 13-34.

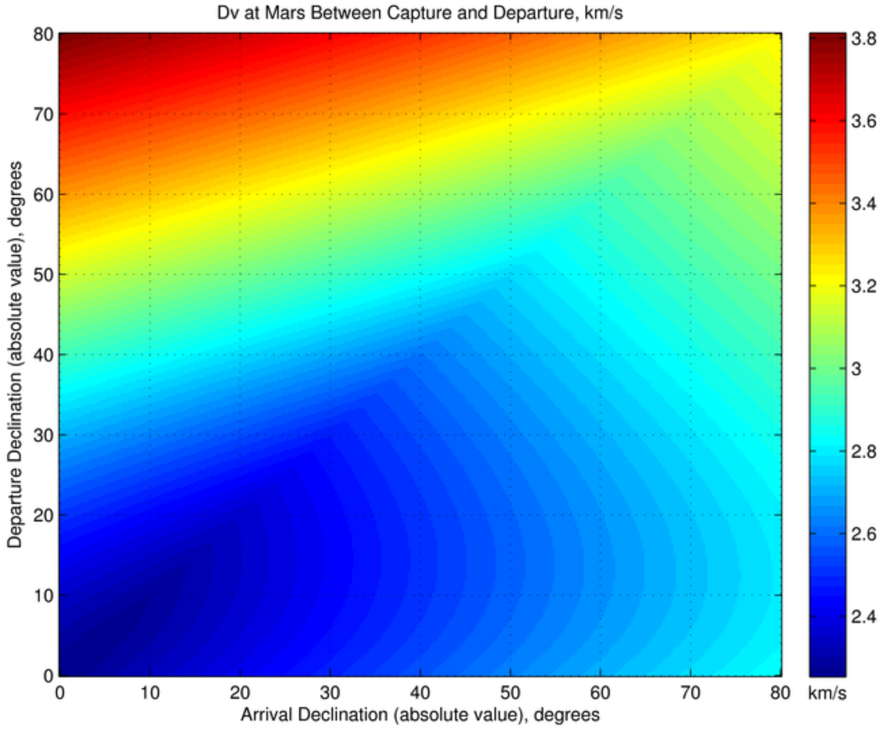


Figure 13-34 ΔV at Mars between arrival (capture) and departure as a function of arrival and departure declinations.

13.1.2.5.5. Full Reorientation of Highly Elliptical Mars Orbits

The results presented thus far address the need for orbit plane changes at Mars solely in terms of arrival/departure asymptotic declinations. However, in practice consideration of the full orientation of the highly elliptical Mars departure orbit such that its inclination, RAAN, and argument of periarieon are all properly selected to contain the outgoing asymptote in the orbit plane and permit the Mars departure maneuver to be performed in the velocity direction at periarieon (to notionally maximize efficiency) must be considered.

An array of results for optimal two-impulse re-orientation of the highly elliptical Mars orbit through various amounts of change in inclination, RAAN, and argument of periarieon to show the associated Δv and flight time requirements is presented. The two-impulse orbit reorientations are optimized for each case using a grid search technique. The array of results is organized as follows: We consider changes in inclination, Δi, of 5^o (Table 13-13), 20^o (Table 13-14), 45^o (Table 13-15), and 70^o (Table 13-16). For each of those changes in inclination computation of the optimal total Δv and flight time for changes in RAAN, ΔΩ, and changes in argument of periarieon, Δω, of 5^o, 15^o, 35^o, 75^o, 180^o, and 300^o was conducted.

The results in Table 13-13 through Table 13-16 indicate that some arrival/departure asymptote orientations at Mars may require up to about 1.5 km/s more reorientation Δv than the 0.1573 km/s that we allocated in Table 13-11 and Table 13-13 for the maneuvering sequence. Note that, as expected, it can be seen that the 0.1573 m/s reorientation Δv in Table 13-14 for Δi = 20^o, ΔΩ = 180^o, and Δω = 180^o (ΔΩ, Δω = 180^o yields the same result as ΔΩ, Δω = 180^o). In addition to requiring extra Δv, the more complicated orbit reorientations also require extra flight time, ranging from several hours to nearly two days.

The majority of these optimal transfers are of one of two basic types. One type, shown in Figure 13-35(a) involves essentially circularizing the elliptical orbit at or near apoarieon, flying to the apoarieon (or nearly so) of the reoriented orbit, and reducing periarieon radius. Some amount of plane change is also included in those two maneuvers. The second type, shown in Figure 13-35(b) involves flying a transfer ellipse that intersects the initial and reoriented orbits at points well prior to (or subsequent to) their apoarieons.

A final point to note is that the Ω and ω of the highly elliptical Mars orbit would both drift under the influence of Mars' non-spherical gravitational field (chiefly due to the J_2 term of the spherical harmonics model for the gravitational field). This means that the Ω and ω to which the capture orbit is reoriented to create the "departure" orbit should be chosen such that those Ω and ω values will naturally drift to their appropriate values for Mars departure after the remaining stay time at Mars has elapsed. For reference, Figure 13-36 shows Ω and ω as functions of inclination for our particular highly elliptical Mars orbit. Unless the orbit is at a relatively high inclination, both Ω and ω will experience non-trivial changes over the course of a several month stay at Mars.

Table 13-13 Total Δv (m/s) / total flight time (hours) required to transfer between 250 x 33813 km altitude Mars orbits with $\Delta i = 5^o$ as a function of ΔΩ and Δω, using minimum Δv two-impulse transfers.

	Δ ω					
ΔΩ	**5°**	**15°**	**35°**	**75°**	**180°**	**300°**
5°	182.5 / 6.3	342.6 / 4.0	547.7 / 44.0	873.7 / 37.3	1293.4 / 20.7	696.6 / 7.7
15°	342.8 / 3.7	474.3 / 45.0	632.0 / 42.3	946.7 / 36.0	1295.3 / 21.7	593.1 / 7.0
35°	547.7 / 43.7	631.7 / 42.3	797.2 / 39.3	1076.9 / 32.7	1257.9 / 18.0	363.4 / 6.0
75°	873.6 / 37.7	946.8 / 36.0	1077.0 / 32.7	1267.4 / 25.3	1088.7 / 12.7	272.8 / 3.7
180°	1293.5 /21.0	1295.4 / 21.7	1258.1 / 17.7	1088.8 / 13.0	39.5 / 24.3	1127.4 / 31.3
300°	696.5 / 8.0	593.2 / 7.0	363.5 / 6.0	272.9 / 3.7	1127.5 / 31.3	1178.9 / 15.3

Table 13-14 Total Δv (m/s) / total flight time (hours) required to transfer between 250 x 33813 km altitude Mars orbits with $\Delta i = 20^o$ as a function of ΔΩ and Δω, using minimum Δv two-impulse transfers.

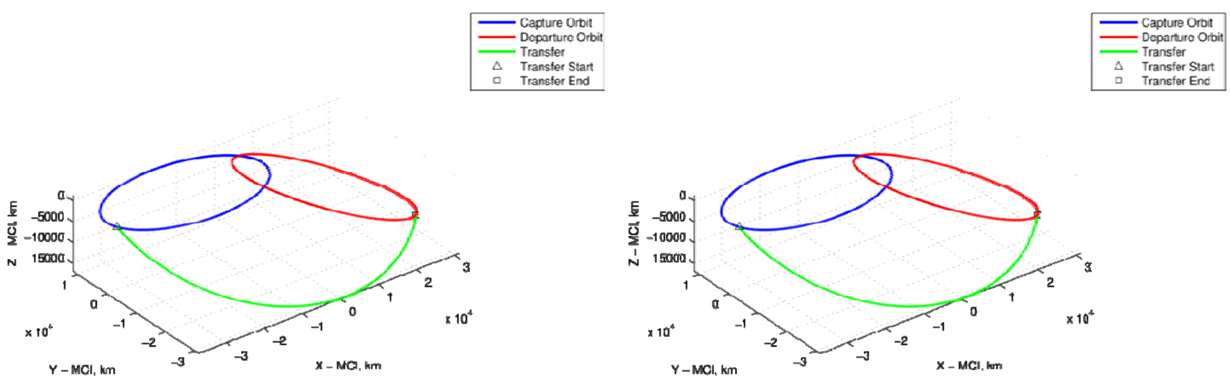
	Δ ω					
ΔΩ	**5°**	**15°**	**35°**	**75°**	**180°**	**300°**
5°	285.3 / 11.3	446.2 / 7.3	695.5 / 41.3	949.7 / 36.3	1321.4 / 20.7	769.7 / 11.0
15°	446.3 / 7.3	625.2 / 5.7	753.2 / 40.3	1023.5 / 34.3	1341.9 / 23.3	674.2 / 10.0
35°	695.3 / 41.7	753.3 / 40.0	889.1 / 37.3	1168.5 / 30.3	1318.5 / 20.3	494.0 / 7.7
75°	949.6 / 36.0	1023.7 / 34.3	1168.4 / 30.3	1453.3 / 21.3	1154.0 / 15.3	567.7 / 5.0
180°	1321.5 / 20.7	1341.8 / 23.3	1318.5 / 20.3	1154.1 / 15.3	157.3 / 1.0	1184.2 / 30.3
300°	769.6 / 11.0	674.2 / 9.7	494.0 / 7.7	567.9 / 5.0	1184.1 / 30.0	1301.6 / 20.0

Table 13-15 Total Δv (m/s) / total flight time (hours) required to transfer between 250 x 33813 km altitude Mars orbits with $\Delta i = 45^o$ as a function of ΔΩ and Δω, using minimum Δv two-impulse transfers.

	Δ ω					
ΔΩ	**5º**	**15º**	**35º**	**75º**	**180º**	**300º**
5º	458.0 / 14.7	615.5 / 9.7	921.3 / 39.0	1132.8 / 34.3	1413.1 / 21.0	934.8 / 13.0
15º	615.3 / 9.7	795.1 / 8.0	976.4 / 37.7	1209.3 / 32.7	1449.7 / 24.0	865.8 / 11.3
35º	921.5 / 39.0	976.4 / 37.7	1092.4 / 35.0	1375.8 / 28.7	1459.2 / 22.3	802.0 / 8.3
75º	1133.1 / 34.7	1209.4 / 32.3	1375.8 / 29.0	1765.5 / 21.3	1310.4 / 17.3	1016.8 / 43.0
180º	1412.2 / 21.3	1449.5 / 23.7	1459.4 / 22.0	1310.5 / 17.7	346.7 / 1.0	1352.7 / 28.7
300º	934.8 / 12.7	865.9 / 11.3	801.9 / 8.3	1017.0 / 43.3	1352.7 / 28.7	1506.8 / 22.3

Table 13-16 Total Δv (m/s) / total flight time (hours) required to transfer between 250 x 33813 km altitude Mars orbits with $\Delta i = 70^o$ as a function of ΔΩ and Δω, using minimum Δv two-impulse transfers.

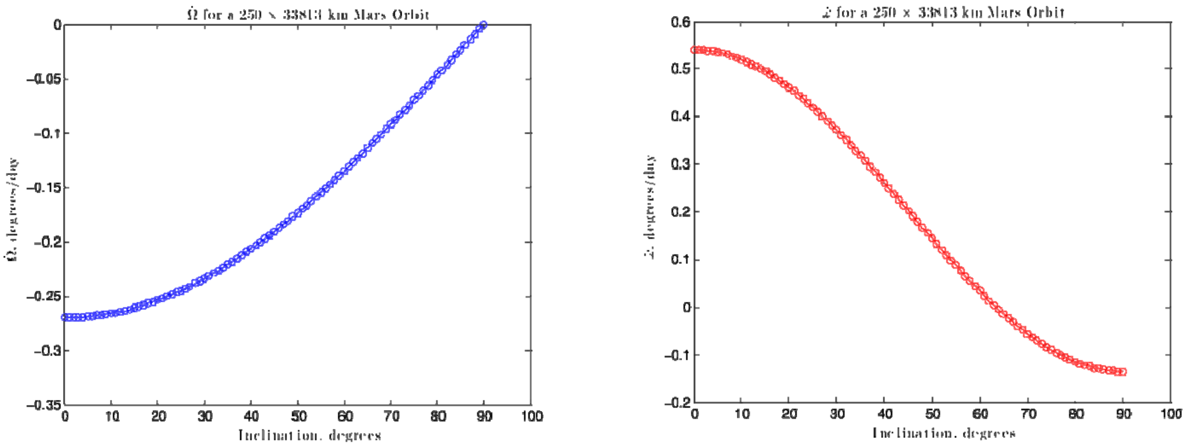
	Δ ω					
ΔΩ	**5º**	**15º**	**35º**	**75º**	**180º**	**300º**
5º	617.0 / 16.0	769.7 / 11.0	1097.6 / 37.7	1305.5 / 34.0	1554.3 / 24.3	1110.4 / 13.3
15º	769.9 / 11.0	936.6 / 9.7	1164.9 / 36.3	1377.8 / 32.3	1595.1 / 23.3	1073.8 / 11.3
35º	1097.6 / 37.7	1164.9 / 36.3	1273.5 / 34.3	1542.0 / 29.3	1627.6 / 22.7	1111.5 / 8.3
75º	1305.6 / 33.7	1378.0 / 32.3	1542.1 / 29.7	1918.7 / 25.3	1483.1 / 18.0	1374.8 / 40.3
180º	1551.8 / 23.3	1594.9 / 24.0	1627.5 / 22.7	1483.1 / 17.7	519.7 / 1.0	1545.7 / 28.3
300º	1110.3 / 13.3	1073.5 / 11.3	1111.7 / 8.3	1374.4 / 40.7	1545.7 / 28.0	1636.8 / 21.3



(a) Optimal two-impulse transfer between highly elliptical Mars orbits with 20º change in inclination, 75º change in RAAN, and 75º in argument of periarieon.

(a) Optimal two-impulse transfer between highly elliptical Mars orbits with 20º change in inclination, 300º change in RAAN, and 15º in argument of periarieon.

Figure 13-35 Example optimal two-impulse transfer trajectories between highly elliptical Mars orbits of different orientations.



(a) $\dot{\Omega}$ as a function of inclination.

(b) $\dot{\omega}$ as a function of inclination.

Figure 13-36 Rates at which the Ω and ω of a 250 x 33813 km altitude Mars orbit will change as a function of orbit inclination due to non-spherical Mars gravity effects (J2).

13.1.2.6. Arrival and Departure Transfers

The incoming (from Earth) and outgoing (to Earth) interplanetary trajectories significantly constrain the geometry of the Mars parking orbit. In general, the planes of the interplanetary trajectories would not coincide with the plane of the target orbit (near equatorial for Phobos and Deimos), requiring additional maneuvers to complete a round-trip mission to Mars [Luidens, 1966][3] [Hoffman, 1991][4] [Desai, 1992][5]. The inclusion of an intermediate parking orbit to stage the Deep Space Vehicle (DSV) further complicates the design of orbital transfers at Mars because parking orbit should allow efficient transfers from both the incoming and outgoing trajectory as well as the target orbit [Cornick, 1970][6] [Cupples, 1993][7][Landau, 2005][8]. Typically, the ΔV of the DSV is kept to a minimum, suggesting low periapsis, long period parking orbits (e.g. 500 km altitude x 1 sol) to minimize the ΔV from the high energy interplanetary trajectories and the captured parking orbit. A less massive Space Exploration Vehicle (SEV) provides the additional ΔV to transfer down to Phobos or Deimos and back to the elliptical parking orbit [Mulqueen, 1986][9] [Foster, 2011][10]. The overall propellant consumption is thus reduced by staging ΔV off the relatively massive DSV onto the smaller SEV. However, efficient orbital transfers are necessary to maintain this mass savings.

13.1.2.6.1. Arrival and Departure Design Techniques

The ΔV to reorient a 500 km altitude x 1 sol parking orbit at Mars is provided in Figure 13-37. The “change line of apsides” (blue and green) lines both perform the same function of rotating the direction of periapsis of elliptical orbits within the plane of the orbit (i.e. no plane change). It would be much more efficient to combine this maneuver with the capture and escape ΔV than to change the argument of periapsis while in the elliptical orbit. Alternatively, it would be more efficient to rotate the orbit along the line of apsides (red line) while the speed is low at the apoapsis of an elliptical orbit, than during the high-speed capture and escape maneuvers. Because these reorientation maneuvers are orthogonal (the blue line changes the line of apsides and does not affect the orbital plane while the red lines changes the orbital plane but does not affect the line of apsides) they can be combined to provide any orbital orientation.

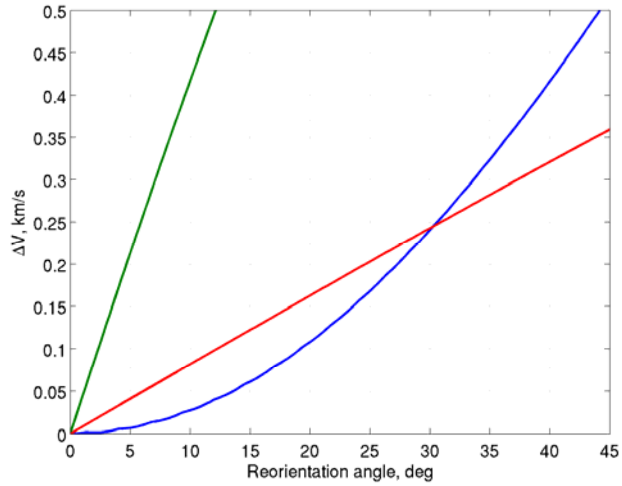


Figure 13-37 Relative cost of orbit reorientation techniques.

Because it is so efficient to rotate along the line of apsides while in orbit, the parking orbits would be designed so that apoapsis remains near the plane of the target orbits. (The orbital plane shifts slightly during the mission due to the oblateness of Mars.) In this way the parking orbit could rotate from the arrival plane to the target plane to the departure plane with efficient maneuvers at apoapsis. However, this technique would require a change in the energy-optimal bending angle during the capture and escape maneuvers to place periapsis (and therefore apoapsis) in the target orbit plane. This additional bending would be most efficiently obtained by altering the capture and escape maneuvers as shown in Figure 1. An entire sequence from capture to Phobos rendezvous to escape is depicted in Figure 13-38.

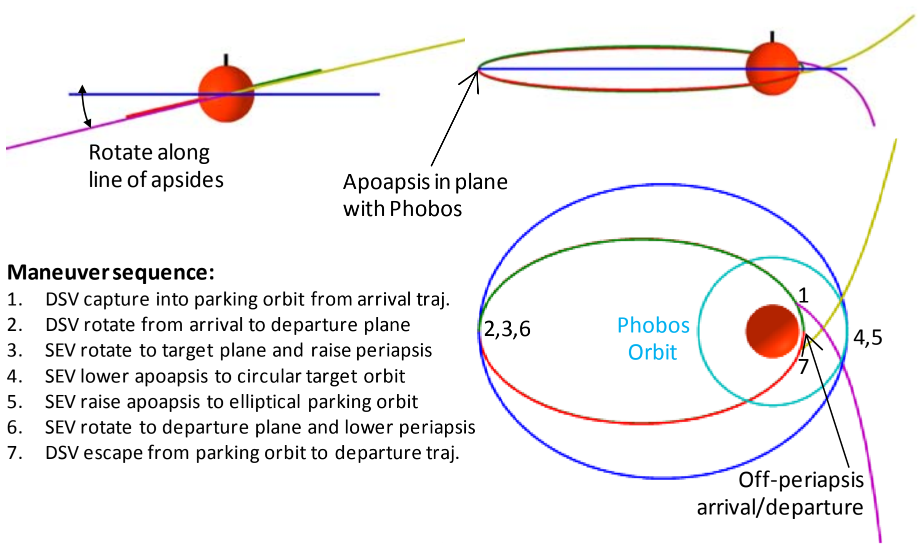


Figure 13-38 Example transfer to and from Phobos.

13.1.2.6.2. Mission Characteristics

The general characteristics of the maneuver sequence depicted in Figure 13-38 are investigated by applying the transfer strategy to short duration (500–800 day round trip) Mars missions. The arrival and departure conditions for opposition class trajectories from 2020 through 2070 are presented in Table 13-17. These V∞ vectors correspond to trajectories designed for relatively low injected mass to low-Earth orbit (including the Earth departure maneuver) and a 60-day stay time at Mars. Although these analyses were conducted utilizing patch-conic methodologies, they provide a good first approximation. Further optimization should be conducted to confirm the results. Table 13-18 contains the ΔV of each maneuver for the DSV at Mars along with the orientation of the parking orbit. The total ΔV for the DSV is found in the second column of Table 13-18, and the portion of the ΔV due to non-ideal geometry is tabulated in the third column. The additional ΔV from the ideal minimum is generally larger than the reorientation maneuver ΔV at apoapsis (column five) because the capture and escape maneuvers (columns four and six) also change the orbital geometry by shifting the line of apsides. The orientation of the parking orbit before and after the reorient maneuver is defined by the right ascension of periapsis (in the target orbit plane) and inclination of the arrival and departure orbits (which share a common line of apsides). The inclinations in Table 13-18 are generally near the equator to provide low-ΔV transfers to Phobos and Deimos. Table 13-19 contains the breakdown of maneuvers for the SEV to transfer to Phobos or Deimos. The total ΔV, as well as the additional ΔV from an ideal geometry (i.e. zero relative inclination), is generally smaller for the SEV than for the DSV, suggesting that most of the non-coplanar ΔV has been placed on the DSV. The maneuver breakdown in Table 13-18 and Table 13-19 is optimized to minimize the sum of the DSV and SEV ΔV, which may not be optimal from a mass standpoint because the DSV could be much more massive than the SEV. An alternative maneuver breakdown is provided in Table 13-20 and Table 13-21 where the ΔV of the DSV is minimized while neglecting the cost of the SEV. In this case the DSV ΔV can be reduced dramatically (e.g. by 800 m/s in 2020) at the expense of an increase in SEV ΔV. The most dramatic changes occur when the inclination of the parking orbit is retrograde and the SEV must nearly reverse direction at apoapsis to transfer to the target orbit. The overall ΔV for the DSV remains low because both the arrival and departure orbits are retrograde. A key design variable for the reorientation ΔV is the speed near apoapsis of the parking orbit, where lower speeds generally lead to lower ΔV. Examples where the apoapsis speed is lowered by increasing the period of the parking orbit to three sols (from one sol) are provided in Table 13-22 and Table 13-23. This design provides the greatest benefit to the SEV ΔV (e.g. over 800 m/s in 2020), though the DSV ΔV also decreases. The SEV ΔV is still generally larger in Table 13-23 than in Table 13-19, but the maneuver combination in Table 13-22 and Table 13-23 could provide an overall lower mass design. However, these longer period orbits provide a transfer opportunity only once every three sols, whereas the shorter period orbits have more frequent transfer opportunities. Thus the mass benefits must be balanced with the operational constraints.

Table 13-17 Mars arrival and departure vectors for short duration (opposition class) missions 2020–2050.

Mars Arrival	Traj. Type[a]	Arrival V_∞, km/s	Arrival RA,[b] deg.	Arrival Dec., Deg.	Departure V_∞, km/s	Departure RA, deg.	Departure Dec., Deg.
12/11/20	EMVE	4.545	124.06	10.98	6.208	45.99	26.69
04/20/23	EME	3.018	-176.10	-18.33	3.198	-85.77	-20.11
06/23/24	EVME	6.712	28.37	10.13	3.322	12.01	7.66
08/11/28	EME	3.239	-11.10	-20.59	2.869	76.74	43.43
10/10/30	EME	3.466	32.23	12.27	2.767	129.83	38.59
01/08/31	EVME	5.427	-151.11	-5.15	3.711	157.19	6.49
10/14/33	EMVE	3.890	5.69	1.23	6.226	0.44	8.51
06/26/35	EVME	5.912	-30.57	-10.76	4.790	-39.57	-15.15
05/16/37	EVME	6.308	-46.45	-26.78	3.872	-15.30	-31.86
06/19/40	EME	2.642	-156.51	-31.08	3.203	-51.50	-4.41
08/08/42	EME	2.709	-117.19	-15.71	3.542	-7.32	9.00
10/02/43	EVME	5.720	138.48	23.66	4.229	78.32	6.04
08/17/46	EMVE	4.400	-44.63	-2.83	6.209	-63.43	-27.30
02/17/50	EME	4.770	149.60	23.19	2.829	-115.72	-12.64
03/19/50	EVME	5.743	-101.06	-25.31	3.761	-125.22	-12.67
12/03/52	EMVE	3.746	95.61	17.72	5.823	50.12	29.04
06/25/56	EVME	5.907	41.70	4.14	3.384	25.99	-5.23
07/05/56	EVME	6.453	27.64	11.12	3.521	14.87	10.52
08/13/60	EME	2.908	-7.10	-20.61	2.675	92.53	43.62
11/01/62	EME	5.348	58.30	25.75	2.891	141.56	31.66
01/10/63	EVME	5.424	-150.39	-6.59	3.402	165.78	2.83
10/16/65	EMVE	3.989	24.39	7.30	6.321	-2.09	11.83
06/28/67	EVME	5.789	-29.71	-9.92	4.734	-35.96	-14.16
04/03/70	EMVE	3.207	174.88	-6.31	4.800	147.76	31.90

a EMVE has a Venus flyby on the return (Mars-Earth) leg; EVME has a Venus flyby on the outbound (Earth-Mars) leg; and EME has no Venus flybys during the round-trip Mars trajectory.

b Mars right ascension and declination are in Mars equator and equinox of J2000 frame.

Table 13-18 DSV transfers for 250 km x 1 sol parking orbits with equal weighting on DSV and SEV ΔV.

	DSV ΔV, km/s					Parking Orbit Orientation[b], deg		
Mars Arrival	Total	Difference from ideal[a]	Capture	Reorient	Escape	Periapsis right asc.	Arrive inc.	Depart inc.
12/11/20	6.47	1.20	2.95	0.06	3.53	-24.8	20.6	28.0
04/20/23	2.32	0.05	1.16	0.00	1.23	141.9	26.3	26.3
06/23/24	5.32	0.42	3.85	0.02	1.52	-73.1	10.3	7.7
08/11/28	2.81	0.60	1.25	0.61	1.04	-61.6	26.0	54.9
10/10/30	2.63	0.34	1.47	0.26	0.98	-9.2	18.2	50.6
01/08/31	5.08	0.96	2.99	0.11	2.04	98.7	5.5	7.6
10/14/33	5.15	0.30	1.75	0.06	3.43	-82.5	1.2	8.6
06/26/35	5.47	0.26	3.18	0.04	2.35	-127.1	10.8	15.2
05/16/37	4.98	0.07	3.40	0.05	1.64	-124.7	27.3	33.4
06/19/40	2.50	0.42	0.93	0.29	1.36	164.8	44.0	7.4
08/08/42	2.88	0.57	1.04	0.33	1.59	-151.1	26.8	15.0
10/02/43	5.72	1.05	3.17	0.14	2.47	28.6	25.0	7.9
08/17/46	5.70	0.53	2.14	0.20	3.45	-141.2	2.8	27.8
02/17/50	3.71	0.55	2.33	0.40	1.09	101.5	29.9	20.4
03/19/50	4.96	0.56	3.11	0.11	1.82	156.4	25.8	12.9
12/03/52	5.23	0.78	2.03	0.08	3.19	-25.1	20.4	29.9
06/25/56	4.79	0.48	3.22	0.08	1.55	-59.1	4.2	5.2
07/05/56	5.16	0.35	3.63	0.01	1.60	-72.5	11.3	10.5
08/13/60	2.57	0.63	1.07	0.65	0.94	-52.4	27.9	58.9
11/01/62	3.77	0.16	2.75	0.07	1.05	7.8	32.0	40.5
01/10/63	4.79	0.85	2.92	0.08	1.84	104.3	6.8	3.2
10/16/65	5.49	0.50	1.96	0.04	3.57	-78.5	7.5	12.2
06/28/67	5.32	0.24	3.08	0.03	2.30	-124.8	10.0	14.2
04/03/70	4.27	0.89	1.56	0.31	2.45	71.8	6.5	32.7

a Minimum ΔV for coplanar transfers.
b Parking orbit orientation is given in Mars equator and equinox of J2000 frame.

Table 13-19 Exploration Vehicle transfers for 250 km x 1 sol parking orbits with equal weighting on DSV and SEV ΔV.

	Phobos SEVΔV, km/s				Deimos SEVΔV, km/s			
Mars Arrival	Total[a]	Difference from ideal[c]	Leave Parking	Return Parking	Total[b]	Difference from ideal[c]	Leave Parking	Return Parking
12/11/20	1.78	0.20	0.30	0.35	1.42	0.15	0.54	0.58
04/20/23	1.81	0.23	0.34	0.34	1.44	0.17	0.57	0.57
06/23/24	1.62	0.03	0.25	0.24	1.29	0.02	0.50	0.50
08/11/28	2.03	0.44	0.34	0.56	1.64	0.37	0.57	0.78
10/10/30	1.94	0.36	0.28	0.53	1.57	0.30	0.53	0.75
01/08/31	1.60	0.02	0.23	0.24	1.28	0.01	0.49	0.50
10/14/33	1.60	0.01	0.23	0.24	1.28	0.01	0.49	0.50
06/26/35	1.65	0.07	0.25	0.27	1.32	0.04	0.51	0.52
05/16/37	1.87	0.28	0.35	0.39	1.49	0.22	0.58	0.62
06/19/40	1.84	0.26	0.48	0.24	1.48	0.21	0.70	0.50
08/08/42	1.74	0.16	0.34	0.27	1.39	0.12	0.58	0.52
10/02/43	1.70	0.12	0.33	0.24	1.35	0.08	0.57	0.50
08/17/46	1.71	0.13	0.23	0.35	1.36	0.09	0.49	0.58
02/17/50	1.79	0.21	0.37	0.30	1.43	0.16	0.59	0.54
03/19/50	1.72	0.14	0.34	0.26	1.37	0.10	0.57	0.51
12/03/52	1.79	0.21	0.30	0.37	1.43	0.16	0.54	0.59
06/25/56	1.59	0.01	0.23	0.23	1.28	0.01	0.49	0.49
07/05/56	1.63	0.05	0.25	0.25	1.30	0.03	0.51	0.50
08/13/60	2.07	0.49	0.35	0.59	1.68	0.41	0.58	0.81
11/01/62	1.96	0.37	0.38	0.45	1.57	0.30	0.61	0.67
01/10/63	1.59	0.01	0.24	0.23	1.28	0.01	0.50	0.49
10/16/65	1.62	0.04	0.24	0.26	1.30	0.03	0.50	0.51
06/28/67	1.64	0.06	0.25	0.27	1.31	0.04	0.50	0.52
04/03/70	1.75	0.17	0.24	0.39	1.40	0.13	0.50	0.61

a Includes 1.13 km/s to enter and depart Phobos orbit.
b Includes 0.29 km/s to enter and depart Phobos orbit.
c Minimum ΔV for coplanar transfers.

Table 13-20 Deep Space Vehicle transfers for 250 km x 1 sol parking orbits with full weighting on DSV ΔV.

	DSV ΔV, km/s					Parking Orbit Orientation[b], deg		
Mars Arrival	Total	Difference from ideal[a]	Capture	Reorient	Escape	Periapsis right asc.	Arrive inc.	Depart inc.
12/11/20	5.61	0.34	2.16	0.13	3.33	169.9	164.9	148.8
04/20/23	2.32	0.05	1.12	0.00	1.20	141.9	26.3	26.3
06/23/24	5.01	0.11	3.70	0.02	1.30	119.8	169.9	172.0
08/11/28	2.80	0.59	1.21	0.59	1.01	-64.1	25.2	56.3
10/10/30	2.63	0.34	1.39	0.28	0.96	-13.6	16.9	53.3
01/08/31	4.24	0.11	2.66	0.10	1.48	-81.8	174.5	172.4
10/14/33	5.15	0.30	1.72	0.06	3.37	-82.5	1.2	8.6
06/26/35	5.47	0.26	3.13	0.03	2.31	-127.1	10.8	15.2
05/16/37	4.98	0.07	3.33	0.05	1.60	-125.7	27.2	33.6
06/19/40	2.49	0.41	0.91	0.32	1.27	169.9	47.4	6.7
08/08/42	2.87	0.57	1.02	0.33	1.53	-149.2	28.0	14.4
10/02/43	4.88	0.20	2.89	0.15	1.84	-157.3	154.0	172.7
08/17/46	5.40	0.23	1.94	0.19	3.28	31.1	177.1	152.6
02/17/50	3.71	0.55	2.27	0.38	1.05	101.7	30.0	20.3
03/19/50	4.54	0.14	2.92	0.09	1.53	-15.9	154.6	166.6
12/03/52	4.53	0.08	1.50	0.07	2.95	154.5	159.5	150.2
06/25/56	4.49	0.19	3.07	0.08	1.35	131.1	175.9	174.6
07/05/56	4.92	0.11	3.50	0.00	1.42	119.4	168.9	169.1
08/13/60	2.57	0.63	1.03	0.63	0.91	-55.1	26.8	60.7
11/01/62	3.77	0.16	2.67	0.08	1.02	5.4	31.2	41.7
01/10/63	4.03	0.08	2.65	0.08	1.30	-75.5	173.2	176.8
10/16/65	5.04	0.05	1.66	0.03	3.35	94.5	172.2	168.1
06/28/67	5.32	0.24	3.03	0.03	2.26	-124.8	10.0	14.2
04/03/70	3.77	0.39	1.22	0.30	2.25	-115.8	173.3	147.9

a Minimum ΔV for coplanar transfers.
b Parking orbit orientation is given in Mars equator and equinox of J2000 frame.

Table 13-21 Space Exploration Vehicle transfers for 250 km x 1 sol parking orbits with full weighting on DSV ΔV.

	Phobos SEV ΔV, km/s				Deimos SEV ΔV, km/s			
Mars Arrival	Total[a]	Difference from ideal[c]	Leave Parking	Return Parking	Total[b]	Difference from ideal[c]	Leave Parking	Return Parking
12/11/20	3.34	1.76	1.12	1.09	3.03	1.76	1.38	1.35
04/20/23	1.81	0.23	0.34	0.34	1.44	0.17	0.57	0.57
06/23/24	3.39	1.80	1.13	1.13	3.07	1.80	1.39	1.39
08/11/28	2.03	0.45	0.33	0.57	1.65	0.37	0.57	0.79
10/10/30	1.96	0.37	0.28	0.55	1.58	0.31	0.53	0.76
01/08/31	3.39	1.81	1.13	1.13	3.08	1.81	1.39	1.39
10/14/33	1.60	0.01	0.23	0.24	1.28	0.01	0.49	0.50
06/26/35	1.65	0.07	0.25	0.27	1.32	0.04	0.51	0.52
05/16/37	1.87	0.28	0.35	0.39	1.49	0.22	0.58	0.62
06/19/40	1.87	0.28	0.50	0.24	1.51	0.23	0.72	0.50
08/08/42	1.75	0.16	0.35	0.27	1.39	0.12	0.58	0.52
10/02/43	3.36	1.78	1.11	1.13	3.05	1.78	1.36	1.39
08/17/46	3.36	1.78	1.13	1.10	3.05	1.78	1.40	1.36
02/17/50	1.79	0.21	0.37	0.30	1.43	0.16	0.60	0.54
03/19/50	3.36	1.78	1.11	1.13	3.04	1.77	1.37	1.39
12/03/52	3.34	1.76	1.12	1.10	3.02	1.75	1.38	1.35
06/25/56	3.39	1.81	1.13	1.13	3.08	1.81	1.39	1.39
07/05/56	3.38	1.80	1.13	1.13	3.07	1.80	1.39	1.39
08/13/60	2.08	0.49	0.34	0.60	1.69	0.42	0.58	0.82
11/01/62	1.96	0.38	0.37	0.46	1.57	0.30	0.60	0.67
01/10/63	3.39	1.81	1.13	1.13	3.08	1.81	1.39	1.39
10/16/65	3.39	1.80	1.13	1.13	3.07	1.80	1.39	1.39
06/28/67	1.64	0.06	0.25	0.27	1.31	0.04	0.50	0.52
04/03/70	3.35	1.77	1.13	1.09	3.03	1.76	1.39	1.35

a Includes 1.13 km/s to enter and depart Phobos orbit.
b Includes 0.29 km/s to enter and depart Phobos orbit.
c Minimum ΔV for coplanar transfers.

Table 13-22 Deep Space Vehicle transfers for 250 km x 3 sol parking orbits with full weighting on DSV ΔV.

	DSV ΔV, km/s					Parking Orbit Orientation[b], deg		
Mars Arrival	Total	Difference from ideal[a]	Capture	Reorient	Escape	Periapsis right asc.	Arrive inc.	Depart inc.
12/11/20	5.34	0.30	2.02	0.07	3.25	173.0	165.6	147.8
04/20/23	2.10	0.06	1.01	0.00	1.08	141.9	26.3	26.3
06/23/24	4.79	0.12	3.59	0.01	1.19	119.8	169.9	172.0
08/11/28	2.26	0.28	1.09	0.28	0.89	-65.6	24.8	57.2
10/10/30	2.24	0.18	1.24	0.15	0.85	-19.1	15.6	57.1
01/08/31	3.95	0.06	2.54	0.05	1.37	-81.7	174.5	172.4
10/14/33	4.82	0.19	1.56	0.03	3.23	87.4	178.8	171.5
06/26/35	5.09	0.11	2.95	0.02	2.12	57.2	169.2	164.7
05/16/37	4.73	0.05	3.22	0.03	1.48	-127.4	27.1	33.9
06/19/40	2.10	0.25	0.82	0.16	1.12	174.5	51.2	6.1
08/08/42	2.52	0.45	0.95	0.16	1.42	-147.7	29.0	14.0
10/02/43	4.58	0.13	2.78	0.07	1.72	-158.7	153.8	172.8
08/17/46	5.08	0.14	1.82	0.09	3.16	31.2	177.1	152.6
02/17/50	3.30	0.38	2.17	0.18	0.95	102.2	30.2	20.1
03/19/50	4.27	0.10	2.81	0.04	1.42	-16.4	154.6	166.6
12/03/52	4.26	0.04	1.38	0.04	2.83	156.1	159.8	150.0
06/25/56	4.23	0.16	2.96	0.04	1.24	131.4	175.9	174.6
07/05/56	4.71	0.13	3.39	0.00	1.31	119.4	168.9	169.1
08/13/60	2.00	0.30	0.91	0.30	0.79	-56.8	26.2	61.8
11/01/62	3.49	0.11	2.52	0.05	0.91	0.5	29.7	44.5
01/10/63	3.75	0.04	2.53	0.04	1.18	-75.4	173.2	176.8
10/16/65	4.80	0.04	1.55	0.02	3.24	94.8	172.3	168.1
06/28/67	4.98	0.14	2.87	0.02	2.10	59.4	170.1	165.8
04/03/70	3.39	0.24	1.11	0.14	2.14	-116.0	173.3	147.9

a Minimum ΔV for coplanar transfers.
b Parking orbit orientation is given in Mars equator and equinox of J2000 frame.

Table 13-23 Space Exploration Vehicle transfers for 250 km x 3 sol parking orbits with full weighting on DSV ΔV.

	Phobos SEVΔV, km/s				Deimos SEVΔV, km/s			
Mars Arrival	Total[a]	Difference from ideal[c]	Leave Parking	Return Parking	Total[b]	Difference from ideal[c]	Leave Parking	Return Parking
12/11/20	2.51	0.83	0.54	0.52	2.03	0.82	0.69	0.67
04/20/23	1.79	0.10	0.17	0.17	1.28	0.07	0.31	0.31
06/23/24	2.53	0.85	0.54	0.54	2.06	0.85	0.70	0.70
08/11/28	1.89	0.21	0.16	0.28	1.38	0.17	0.31	0.41
10/10/30	1.87	0.18	0.14	0.28	1.36	0.15	0.29	0.41
01/08/31	2.53	0.85	0.54	0.54	2.06	0.85	0.70	0.70
10/14/33	2.53	0.85	0.54	0.54	2.06	0.85	0.70	0.70
06/26/35	2.53	0.84	0.54	0.54	2.05	0.84	0.69	0.69
05/16/37	1.81	0.13	0.17	0.19	1.31	0.10	0.31	0.33
06/19/40	1.83	0.14	0.26	0.12	1.33	0.12	0.39	0.27
08/08/42	1.76	0.08	0.18	0.13	1.27	0.05	0.32	0.28
10/02/43	2.52	0.84	0.53	0.54	2.04	0.83	0.68	0.70
08/17/46	2.52	0.83	0.54	0.53	2.04	0.83	0.70	0.68
02/17/50	1.78	0.10	0.18	0.15	1.28	0.07	0.32	0.29
03/19/50	2.52	0.83	0.53	0.54	2.04	0.83	0.68	0.69
12/03/52	2.51	0.82	0.53	0.52	2.03	0.82	0.69	0.68
06/25/56	2.53	0.85	0.54	0.54	2.06	0.85	0.70	0.70
07/05/56	2.53	0.84	0.54	0.54	2.05	0.84	0.69	0.69
08/13/60	1.91	0.23	0.17	0.30	1.40	0.19	0.31	0.43
11/01/62	1.86	0.18	0.18	0.23	1.35	0.14	0.32	0.37
01/10/63	2.53	0.85	0.54	0.54	2.06	0.85	0.70	0.70
10/16/65	2.53	0.85	0.54	0.54	2.06	0.84	0.70	0.69
06/28/67	2.53	0.84	0.54	0.54	2.05	0.84	0.70	0.69
04/03/70	2.51	0.83	0.54	0.52	2.04	0.83	0.70	0.67

a Includes 1.45 km/s to enter and depart Phobos orbit.
b Includes 0.67 km/s to enter and depart Phobos orbit.
c Minimum ΔV for coplanar transfers.

13.1.2.7. Exploration of Phobos and Deimos Mission Design Conclusion

In this paper we have studied the precise orbital motion of Phobos and Deimos about Mars and used that data to inform the design of optimal rendezvous trajectories between Phobos and Deimos. We extend those results to construct complete maneuvering sequences for exploring Phobos and Deimos, including arrival at Mars from hyperbolic approach, rendezvous with each moon, proximity operations at each moon, and Mars departure. We applied this notional maneuvering sequence across a set of round-trip Mars trajectory solutions obtained from trajectory scans that include opposition class trajectories (with and without Venus gravity assists) and conjunction class trajectories to assess requirements for total Δv at Mars for Phobos/Deimos exploration missions. Finally, we assessed the effect on total Δv at Mars due to the reorientation of highly elliptical Mars orbits that may be required to properly align with incoming/outgoing asymptotes at Mars.

Our results indicate that the total Δv at Mars for Phobos/Deimos exploration will likely range between 4.8 to 9 km/s, depending on the mission's departure date and type of round-trip trajectory own. The total time required for orbital maneuvering to explore Phobos and Deimos will likely range between 1.5 to 2 weeks depending on the exact conditions encountered. This leaves ample time to spend exploring the moons themselves provided that the stay time at Mars is at least several months.

13.1.2.7.1. Future Work

One of the next steps in the analysis will be to implement the maneuvering sequence in a high-fidelity end-to-end simulation environment to fully validate the maneuver sequence, develop exemplar point designs using precision integrated trajectories and finite burn maneuvers, and then construct detailed design reference missions using those results.

also It is also desired to further study the problem of proximity operations at Phobos and Deimos, including identification of robust and safe strategies for spacecraft proximity operations at each moon and for maneuvers with which the crew can interact with the surfaces of the moons. Preliminary results indicate that Phobos may be more challenging than Deimos in this regard. Future studies of stable periodic motion in the vicinity of the moons using complete force models will inform these analyses. Other issues include the refinement of the terminal rendezvous sequences to account for relative navigation sensor and filter performances in the presence of realistic environmental factors such as lighting conditions. The results of such analysis will likely lead to modifications of the notional terminal rendezvous sequences presented herein.

13.2. Mars Program Planning Group Summary

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13.2.1. Mars Program Planning Group Charter

The President's FY2013 Budget Request contained reductions in the Mars future program line necessitating reformulation of the current Mars Exploration Program (MEP), and discontinuing development of 2016 and 2018 missions with the European Space Agency. The realities of the fiscal environment, new priorities, and the most recent inputs from the science, human exploration, and technology communities, provided an opportunity to set new directions and a revised vision aimed at revising and renewing the program. This opportunity also served to exploit synergies between NASA programs to take advantage of the strengths of the NASA robotic and human exploration efforts for the long-term future (2025 and beyond).

The NASA Administrator directed the Associate Administrator for the Science Mission Directorate (AA/SMD) to lead Mars program reformulation activities working with the Associate Administrator for Human Exploration and Operations Directorate (HEOMD), the NASA Chief Technologist (OCT), and the NASA Chief Scientist (OCS). In support of this reformulation, NASA established a Mars Program Planning Group (MPPG)[5]. The purpose of the MPPG was to develop foundations for a program-level architecture for robotic exploration of Mars that is consistent with the President's challenge of sending humans to Mars orbit in the decade of the 2030s [Obama, 2010][11], yet remain responsive to the primary scientific goals of the 2011 NRC Decadal Survey for Planetary Science [NRC, 2011][12]. Program architecture was defined as a sequence of strategically selected and interconnected spaceflight and ground-based investigations that would increase scientific knowledge, advance key technologies, and inform and would enable long-term human exploration goals. The MPPG was structured to serve as a limited-term study group responsible for delivering specific products to aid NASA in the decision-making process on the future direction of the reformulated MEP.

The immediate focus of the MPPG was on the collection of multiple mission concept options for the 2018/2020 Mars launch opportunities. Fidelity and timeliness of these studies were the MPPG's priority so as to affect Agency decisions in the upcoming FY14 budget planning process and ensure a 2018 mission would be viable. To maintain the successful strategic structure of the MEP, and ensure relevancy of the possible 2018/2020 mission(s) to longer-term science and exploration priorities, notional architectures/pathways spanning to the 2030s were required, including future human exploration of Mars. The purpose of the MPPG was to develop foundations for a program-level architecture for robotic exploration of Mars that is consistent with the President's challenge of sending humans to Mars in the decade of the 2030s, yet remain responsive to the primary scientific goals of the 2011 NRC Decadal Survey for Planetary Science. Consistent with its charter, MPPG reached out to internal and external science, technology and engineering communities, to develop mission options and program architecture alternatives for NASA's consideration. The MPPG was specifically chartered to provide options that integrate science, human exploration and technology at an Agency level with Mars Exploration as a common objective.

13.2.2. Human Exploration beyond LEO: A Capability Driven Framework

The option of a crew returning Mars surface samples which had been previously placed into Mars orbit via a robotic

[5] Additional information regarding the MPPG can be found at: http://www.nasa.gov/offices/marsplanning/home/index.html.

mission was particularly intriguing during initial deliberations of the MPPG core team, especially with its direct ties to the National Space Policy [Office of the President, 2010]. This line of thinking spawned a series of questions regarding the potential of humans returning samples from Mars orbit – a non-landed mission to Mars. In order to understand the viability of this strategy, assessments of the implications of both the development schedule, including technology development, as well as the risk were required. Previous assessments have determined that a mission to Mars as the first destination for humans beyond Earth orbit may not be the best strategy [NASA, 1989][13], [Synthesis, 1991][14], [Augustine, 2009] to name a few. Rather, in order to establish a balanced risk/cost posture, a progressive expansion of humans from Earth, with near-Earth destinations as the initial missions was best warranted and thus a less destination specific framework has emerged. This strategy, referred to as a Capability Driven Framework (CDF) [NASA, 2011][15], is based on the idea of an ever expanding human presence beyond low-Earth orbit in terms of duration and distance from the Earth. It is based on evolving capabilities which would be utilized after operational experience had been established from completing less demanding missions. In theory, the CDF would enable multiple destinations and would provide increased flexibility, greater cost effectiveness, and sustainability. But the utility of a CDF can only be measured and fully understood when put into context of actual missions. Thus, to help formulate the strategies, technologies, and systems needed to support the framework, example destinations are being examined including low-Earth orbit, Geostationary missions, cis-lunar space (including lunar fly-by, lunar orbit, and lunar surface), Near-Earth Asteroids (NEA), as well as missions to the Mars and the moons of Mars. Before examining how human missions to Mars would fit into the overall CDF a brief review of the missions associated with the CDF is necessary.

Geostationary Orbits (GEO): This mission class includes missions to GEO or other high-Earth orbit destinations generally for the purpose of deploying or repairing ailing spacecraft. Due to the high change in velocity (ΔV) associated with these destinations; a split-mission approach would be typically used where the crew would be sent to the destination separate from the cargo assets to be used at the destination. The cargo assets could include habitats, mobility systems, robotic systems, and repair equipment.

Earth-Moon Libration: This mission class includes missions to the Earth-Moon libration points (L1 or L2) or high lunar orbit. As with the GEO mission, cargo for these missions would be typically sent separately from the crew. L1/L2 could also serve as a staging node for other destinations such as to the lunar surface, NEAs, or perhaps even Mars. Thus, crew missions to L1/L2 may serve as the initial crew transport leg at the beginning or end of a different mission class.

Lunar Surface: Missions to the lunar surface would encompass a range of mission durations, beginning with short stays to prove the performance of the systems, to longer duration test beds for more challenging missions such as the surface of Mars. As with both the GEO and L1/L2 missions, a split mission approach would be typically used separating the crew from cargo.

Near-Earth Asteroids: This mission class represents human missions to and from asteroids which are in close proximity to Earth, orbit perihelion typically less than 1.3 Astronomical Unit. Near-Earth Asteroids are of interest because they represent a class of missions which truly leave Earth vicinity. Since these missions would be conducted in heliocentric space and the orbits of least energetic NEAs have long synodic periods, perhaps decades long, it becomes difficult to pre-deploy mission assets at the NEA prior to the crew mission. Thus, these missions would be typically constructed as all-up missions, whereby all of the required mission assets would be transported with the crew (deep space habitat, destination exploration systems, and Earth entry vehicle).

Mars Orbit: This mission class includes missions to the moons of Mars (Phobos and Deimos) as well as Mars orbit. Missions to Mars can occur approximately every 26 months due to the difference in the orbital periods of Earth and Mars. Since these missions avoid planetary surfaces, the crew would be exposed to the deep space environment for the entire mission duration. Thus, these missions are generally constructed to reduce this crew exposure by flying the trajectories as fast as possible within the constraints of the propulsion technologies and number of heavy lift launches. Since missions to Mars could occur on a frequent basis (every 26 months), pre-deployment of mission exploration vehicles is usually employed.

Figure 13-39 provides an example mission profile for a typical Mars orbital mission. With this mission construct cargo, which would be utilized to explore the Mars system when in orbit, would be pre-deployed to Mars one opportunity before the crew would depart Earth. This strategy would allow the cargo to utilize energy-efficient trajectories, thus reducing the propellant mass and resulting architecture mass. It would also allow for operational checkout of the cargo to ensure that it arrived safely and operating as would be expected before the crew mission. The number of launches required is dependent on the transportation technology used and the payload deployed.

The Mars moon exploration concept envisioned here would use a large interplanetary spacecraft to transport a crew to and from Mars. Upon arrival at Mars, it would be placed into a high Mars orbit. Upon arrival, the crew vehicle would rendezvous with the pre-deployed cargo placed in this parking orbit on a previous transfer opportunity. Half of the crew would use a Space Exploration Vehicle (SEV) and one of three chemical orbit transfer stages (OTS) to transfer from this parking orbit to the vicinity of Phobos and spend the next two weeks exploring this moon. Because the orbits of both Phobos and Deimos are nearly in the equatorial plane of Mars and the arrival and departure declinations of the transfer trajectories do not typically lie in that same plane, large plane change maneuvers are required to transfer the crews from the high parking orbit down to the orbits of the moons. After returning from this Phobos mission, the other two crew members would use a second SEV and OTS to transfer from the parking orbit to the vicinity of Deimos and spend the next two weeks exploring the other Martian moon. A third OTS would be available to rescue either crew should they become stranded at either Phobos or Deimos. Crew time not used to explore Phobos or Deimos would be available to potentially retrieve samples from a separate robotic sample return mission or perhaps teleoperate robotic systems on the surface of Mars when a communication path is available. At the end of the orbital stay[6], all SEV and OTS assets would be jettisoned, and the large interplanetary spacecraft would depart from its parking orbit for the return trip to Earth.

Mars Surface: This mission class represents missions to the surface of Mars. Strategies for exploring the surface of Mars typically utilize pre-deployed cargo vehicles and flying lower energy conjunction class missions. For the surface long-stay mission class, the NASA Design Reference Architecture 5.0 [Drake, 2009] was utilized as the basis for the analysis. For this mission, a crew would be sent to Mars on a long-stay class trajectory. Upon arrival the crew would place their large interplanetary vehicle into a high-Mars parking orbit to rendezvous with one of two cargo vehicles sent to Mars on the prior orbit transfer opportunity. The second cargo vehicle would have already landed at the intended surface exploration location, where automated systems would have set up a power plant and a propellant manufacturing plant. When all necessary systems have been verified operational and landing conditions determined to be satisfactory, the crew would initiate the landing sequence. Once the crew landed at this site, they would spend approximately 500 days exploring the vicinity in a series of long traverses (several 100 kilometer) extending from this fixed central base – an approach dubbed the “commuter” strategy. At the completion of this surface mission, the crew would ascend from their surface base, using propellants manufactured there, and return to the waiting interplanetary vehicle. At the appropriate time the crew would depart from Mars for a six-month transfer back to Earth. Figure 13-40 provides an example mission profile for a typical Mars surface mission.

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[6] Because there are two distinct trajectory types for a round-trip mission to Mars, there are also two distinct approaches for potentially conducting exploration missions while in Mars orbit. Short-stay opposition class mass mission can reduce the total mission duration by flying the trajectories as fast as possible. These missions are generally 600-800 days in duration with approximately 60 days at Mars. On the other hand, using the longer duration trajectories (i.e., the long-stay class with 500 days at Mars) would reduce the number of launches (for a fixed payload mass) but at the expense of increasing a crew’s exposure to the deep-space zero-gravity and radiation environment (up to 1000 days).

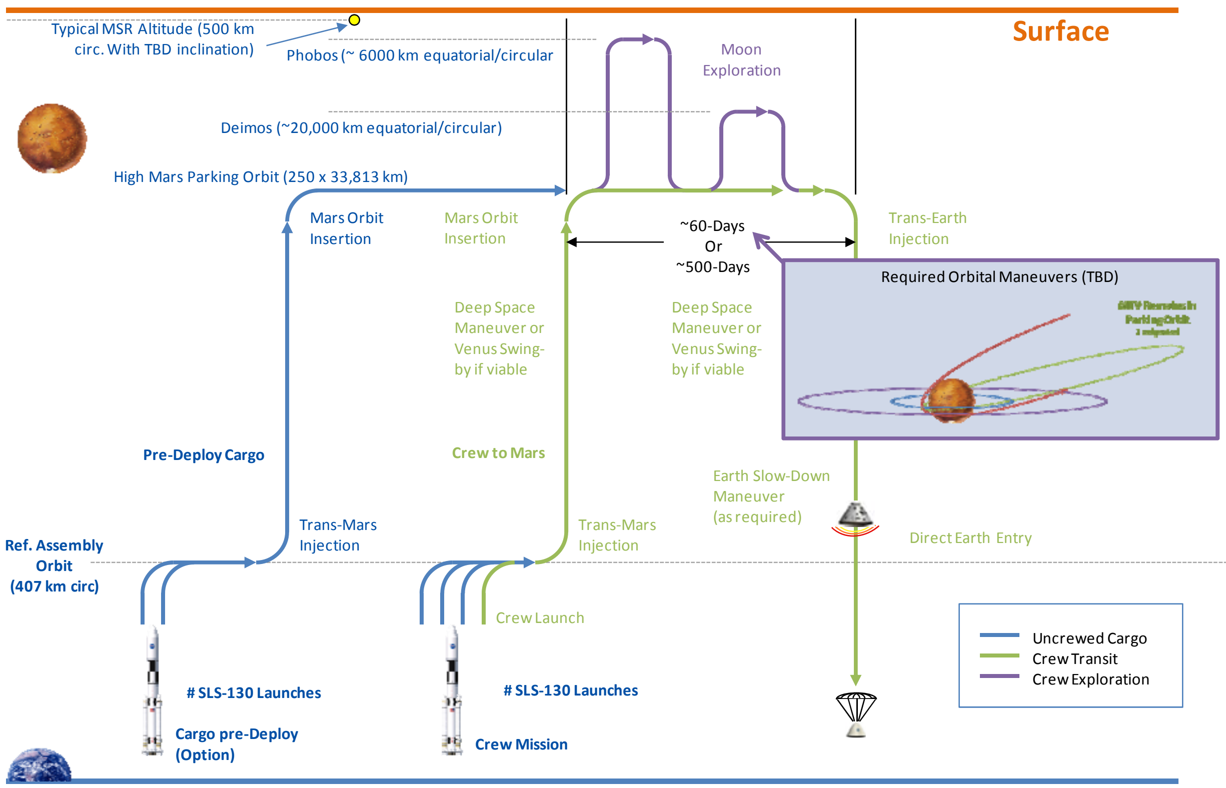

Figure 13-39 Typical mission profile for a Mars orbital mission.

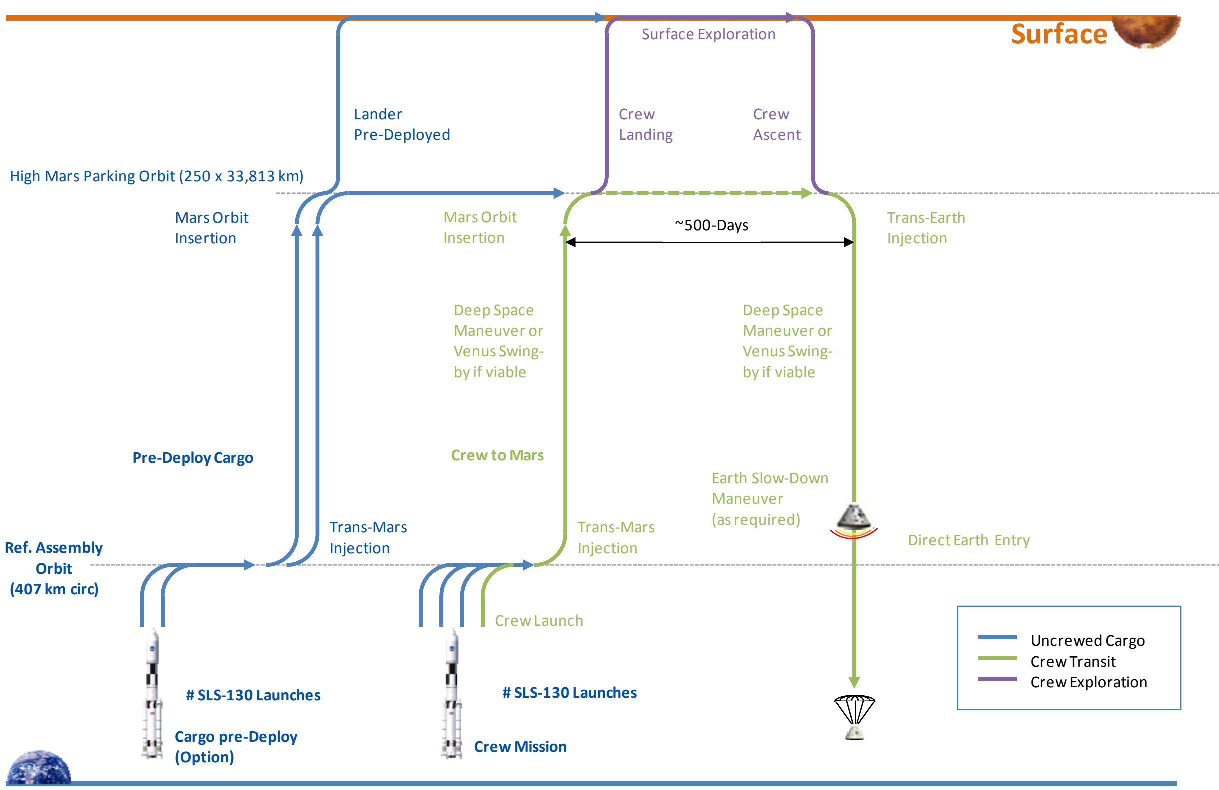

Figure 13-40 Typical mission profile for a Mars surface mission.

13.2.3. Humans to Mars Risk Assessments

One of the first tasks given to the Human Spaceflight Architecture Team in support of the MPPG was to provide a high level technical and schedule feasibility assessment of a crewed visit to Mars orbit, with a launch in 2033, to retrieve samples which had been previously placed into Mars orbit robotically and then subsequently return them to Earth. The team was specifically requested to identify, while making reasonable assumptions, strategic roadmaps for future human exploration of Mars. The following attributes of human exploration of Mars were to be considered when generating those roadmaps:

- The fundamental technologies, knowledge needed and key options
- The decision points at which technology use and down-select must be made
- Those decisions and deliverables to and from the technology office and, or the science directorate
- The risk postures associated with options

Before detailed roadmaps could be generated, further understanding of the key challenges and risks are required. Mitigation strategies for these risks will compose the primary content of the decision paths, options, technologies, and resulting implementation content.

13.2.4. Human Support Risks

As humans venture farther and longer into deep-space it is necessary that informed risk decision are made in terms of how to best support these explorers. The Human Research Program (HRP) was formed to focus NASA's research on the highest risks to human health and performance during exploration missions. The HRP performs research necessary to understand and reduce spaceflight human health and performance risks in support of exploration, develop technologies to reduce medical risks, and to develop human spaceflight medical standards for subsequent system development and design. The HRP content has been organized to address the key technological challenges of long duration spaceflight:

- Space Radiation: Human health effects, limiting factors for vehicle environments and crew selection; computational shielding modeling; measurement and warning technologies
- Exploration Medical Capability: Medical care and crew health maintenance technologies (monitoring, diagnostic, treatment tools and techniques); medical data management; probabilistic risk assessment
- Human Health Countermeasures: Integrated physiological, pharmacological and nutritional countermeasures suite; Extra-Vehicular Activity (EVA) related physiology research to support new EVA suit development
- Behavioral Health & Performance: Behavioral health selection, assessment, and training capabilities; intervention and communication techniques to support exploration missions
- Space Human Factors & Habitability: Anthropometry, display/control, usability, cognition, habitability, lighting, ergonomics; advanced food development; lunar dust characterization and toxicological testing
- ISS Medical Project: ISS research integration and operations
- National Space Biomedical Research Institute (NSBRI): Nationally competed/peer-reviewed research projects addressing above content utilizing investigators at more than 70 institutions in 22 states

The HRP follows an evidence- and risk-based management approach where the validity of risks is based on evidence from science, clinical, and operational research. Risks are externally reviewed by Institute of Medicine (IOM) review validated HRP exploration risks and evidence. Throughout the HRP process continuous evaluation of risk postures and priorities are assessed and re-evaluated annually based on the current research results and progress. Criticality and priorities are developed in conjunction with Human System Risk Board. Through this process gaps and research tasks are prioritized balancing customer need (from Program elements, Chief Medical Officer, Medical Operations) with flight and ground resources, including ISS availability.

The HRP currently uses three mission classes to inform the risk-based processes including six month lunar outpost missions, a notional 12 month mission to a near-Earth asteroid, and a 30 month Mars landing consistent with the Mars Design Reference Architecture 5.0. The HRP has a 'criticality rating' for 31 human health and performance risks relevant to a DRA 5 type Mars mission. These criticality ratings are characterized as:

- “Unacceptable” = HRP would recommend “No-Go” today
- “Acceptable” = HRP would recommend “Go” with reservations today
- “Controlled” = HRP would recommend “Go” without significant reservations today

These criticality rating are guided primarily by likelihood and consequence scores in a risk management 5x5 (likelihood x consequence) matrix. For this MPPG activity results from previous HRP assessments were used to inform the humans to Mars orbit strategic questions in order to provide guidance as to the human support risk posture associated with an orbit only mission mode. Table 13-24 provides a comparison of the estimated criticality ratings for a notional surface (aka DRA 5) and Mars orbital (600-900 day) mission. It should be noted that these are estimates only based on what we know today as extrapolated from the current knowledge base to these longer-duration missions. As further research is obtained from missions to the ISS or other destinations it is anticipated that these risks will become more controlled and subsequently “acceptable”. But a great deal of research and analysis are required to close the current HRP risks.

As can be seen from Table 13-24 it is believed that the risk to the human system is higher for an orbit-only mission as compared to a surface landing mission[7]. From this preliminary assessment it was determined that 23 risks increase for an orbital-only mission whereas four risks are unchanged and four risks decrease. Since the crew for orbital missions are confined to limited volumetric spaces in the transit habitat for very long durations (600 to 900 days) two additional risks might become unacceptable for the orbital missions including team cohesion and human factors/vehicle habitat design. Due to the uncertainty and lack of detailed research data related to these very long zero-g missions, there was not a major qualitative difference in the risk to the human system when comparing a 600 day opposition class and 900 day conjunction class missions. Both are considered long, in a deep-space zero-gravity environment and well beyond our current experience base. A key strategy for reducing these risks identified by the HRP team is to obtain relevant human system performance data via sequential deep space exposures of 30, 60, 180, and 360 days which would permit gradual testing of radiation effects, behavior, and habitability beyond the protective environment of Earth.

13.2.5. Key Challenges for Orbital and Surface Missions

In addition to the human support risks, other aspects of the mission risk including both loss of crew, as well as loss of mission, must be considered in the roadmap framing. A high-level risk modeling effort was initiated to support the risk assessment for both the orbital and surface missions. This risk modeling exercise stemmed from the assessment methodology established during framing of DRA 5.0. Further discussion of the risk modeling is provided in later sections of this Addendum.

Due to time and resource limitation the integrated architecture risk model is currently at a very high level. Since the risk modeling process was essentially stopped after DRA 5.0 and it is currently being re-established, sufficient time and resources have not be available to develop detailed models. Thus, these initial findings represent rough first cut estimates should not be viewed as absolute predictions. The data comprising the integrated model are chosen from variety of sources for “best fit” including the International Space Station, Shuttle, Constellation (typically derived from ISS or SSP), other space systems (such as satellites or launch vehicles), as well as other related analyses (crew medical, radiation, etc.). At this early stage of risk modeling the intent was not to determine absolute risk values, but rather determine the key risk drivers for the various missions, mission phases, and elements. A comparison of the key challenges for the notional orbital and surface (DRA 5) missions is provided in Figure 13-41. Examination of Figure 13-41 indicates that orbital missions result in more challenging missions with respect to supporting humans in deep-space as well as the overall in-space transportation technologies and architectures. Since human support challenges for orbit only missions is increased, more emphasis is placed on transportation system performance to reduce the mission duration which also increases the total mission mass. Surface missions, on the other hand, generally contain more programmatic challenges, due to the additional number of vehicles and systems, as well as the challenges of entry, descent, landing and ascent.

Once the key challenges have been identified, mitigation strategies for the risk drivers can be determined. Table

[7] It should be noted that this finding is specific to the human system only. That is, other mission related risks, such as entry, descent, landing, ascent, and surface operations are other risks that are not directly related to the human system, but are risks that must be adequately addressed.

13-25 provides an example of the identified key risk drivers for future human missions to Mars.

Table 13-24 Human support challenges.

HRP Risk	Mars DRA5	Mars Orbit	Duration comments wrt DRA5 baseline
Visual impairment	U	↑	↑time in weightlessness
Behavioral health	U	↑	↑time in ICE
Muscle	U	↑	↑time in weightlessness [1]
Aerobic capacity	U	↑	↑time in weightlessness [1]
Radiation- degenerative	U	↑	↑radiation exposure [2]
Radiation- cancer	U	↑	↑radiation exposure [2]
Nutrition	U	↔	assumes no Mars crops
Food	U	↔	assumes no Mars crops
Medical care	U	↓	no planetary EVA
Human Factors- Vehicle/Habitat	A	↑	↑time in ICE & Behavioral Health
Team cohesion	A	↑	↑time in ICE & Behavioral Health
Spacecraft control & egress	A	↑	moon ops; ↑time in ICE (↓cognition)
Radiation- CNS	A	↑	↑radiation exposure [2]
Radiation- Acute (SPE)	A	↑	↑radiation exposure [2]
Human Factors- Task design	A	↑	↑time in ICE (↓cognition)
Human Factors- Training	A	↑	↑time in ICE (↓cognition)
Human Factors- Robotics	A	↑	↑time in ICE (↓cognition)
Human Factors- Computers	A	↑	↑time in ICE (↓cognition)
Immune	A	↑	↑time in ICE
Host-microorganism	A	↑	↑time in ICE
Orthostatic intolerance	A	↔	↑time in weightlessness [1]
Cardiac arrythmia	A	↔	↑time in weightlessness [1]
Intervertebral disc	A	↔	↑time in weightlessness [1]
Osteoporosis	A	↔	↑time in weightlessness [1]
EVA health and performance	A	↔	weightless geology
Medications	A	↔	↓mission time & drug stability
Dynamic loads	I	↔	↑time in weightlessness [1]
Kidney stones	C	↔	↑time in weightlessness [1]
Dust or volatile exposure	n.d.	↓	no exposure in DSH
Fatigue	C	↓	no circadian entrainment
Bone fracture	C	↓	no falling in weightlessness

U	Unacceptable risk that would keep a mission from proceeding
A	Acceptable as is, but with a high uncertainty in risk; additional mitigation recommended
C	acceptable through use of known Controls
I	Risk is poorly understood or no standard exists; Insufficient data
↔	No anticipated change in trend
↑	Anticipated to trend worse
↓	Anticipated to trend better

[1] criticality rating will probably be reduced in July 2012 due to bone/muscle/cardio countermeasure development

[2] details depend strongly on trajectories and proximity to Mars and moons

	Mars Orbit*	Mars Surface
Human Health and Performance		
Time in zero-gravity free space (days)	600-900	180/180
Time on surface (days)	0	540
Galactic Cosmic Radiation Protection	✓✓	✓
Behavioral Health for 600-900 days	✓✓	✓
Key Capability Gaps		
130 t SLS Launch, Large volume, campaign	✓✓	✓
Orion 900 day dormancy, 6 crew	✓	✓
900 Day class deep-space habitation	✓✓	✓
Advance in-space propulsion (e.g. NTP, NEP)	✓✓	✓
20-40 mt (payload) lander	n/a	✓
30 kW-class continuous fission surface power	n/a	DRA 5
Technology Development		
Aerocapture	✓	DRA 5
Automated Rendezvous and Docking	✓	✓
Zero-boiloff cryogenic propulsion	✓	✓
Mars Ascent Methane-oxygen Propulsion	n/a	✓
High Speed Earth Entry	✓✓	✓
Atmospheric based ISRU	n/a	DRA 5
System Reliability	✓	✓
High Reliability Closed Life Support	✓	✓

	Mars Orbit	Mars Surface
Key Precursor Knowledge		
Atmosphere Dynamics	✓	✓✓
Surface Material Properties	✓	✓✓
Planetary Protection	✓	✓✓
Mission Mode (Short/Long Stay)	✓	DRA 5
Pre-deployed Mission Cargo	✓	DRA 5
ISRU for Ascent from Surface	n/a	✓
Destination Exploration Operational Concept	✓	✓
Launch Campaign and Launch Availability	✓✓	✓
Integration and Programmatic		
Integrated Strategy / Plan	✓	✓
Multiple large-scale Technology Programs	✓	✓✓
Multiple Concurrent System Developments	✓	✓✓
Infrastructure Investments	✓	✓
Subscale Demonstrations	✓	✓✓
Continuity of Multiple Developments	✓	✓✓

*	Assumes opposition class (short-stay) missions implemented to reduce crew exposure to the deep space environment
✓	Key challenge applicable
✓✓	Increasing difficulty of key challenge area
DRA 5	Represents agency decisions per Mars Design Reference Architecture 5.0 (NASA-SP-2009-566)
n/a	Not applicable

Figure 13-41 Key challenges and risks for future human exploration of Mars.

Table 13-25 Example risk mitigation venues for top exploration risks.

	Risk Mitigation Venue				
Top Risk Area	**Earth**	**ISS**	**Cis-Lunar**	**Deep Space**	**Mars Robotic**
Reliability of spacecraft hardware	✓	✓	✓	✓	
Human support	✓	✓	✓	✓	
Orion crew vehicle reliability	✓		✓	✓	
Entry, descent, and landing at Mars	✓				✓
ISRU and Mars ascent	✓				✓
Advanced propulsion systems	✓			✓	

13.2.6. Humans to Mars Roadmap Observations

Because of the uncertainty associated with the anticipated NASA budget which may be available for a future human to Mars are not known and difficult to anticipate, the assessments were conducted without specific budgetary assumptions applied. That is, the schedules and implementation options developed were done assuming that the associated required budget was a free variable. For this schedule assessment, a mission consisting of humans to Mars orbit was used as the primary emphasis, which was consistent with the original MPPG thinking and guidance, with humans to the Mars surface as a secondary emphasis. Understanding the linkages of the latter, surface missions, was necessary for strategic guidance. That is, humans to Mars orbit should feed forward to surface missions. It should be noted that for this exercise a notional mission to Mars orbit was assumed to occur in 2033 consistent with the current National Space Policy [Office of the President, 2010]. For this schedule exercise it was further assumed that a surface mission would occur two opportunities (approximately 4 years) later. This was assumed to ensure that core competencies and capabilities developed for the orbital mission could feed forward to the surface mission. As the gap between missions increase, it was felt that the ability to maintain those core competencies would erode, thus leading to additional technical and programmatic risk. To constrain the planning three different flavors of development "content" were constructed, each of which were developed consistent with differing levels of relative development risk.

- *Nominal*: With this development approach it was assumed that all due diligence would be done in the design, development, and testing to have an acceptable level of risk. Margin would be allocated and managed according to typical space development efforts.
- *Aggressive*: With this approach no limitations on funding were assumed for the technology and systems development which were further assumed to be pursued aggressively. In theory, this schedule approach would demonstrate how fast the product could be developed and ready for use. Basically crash the schedule and find the critical path, identifying the limiting schedule constraints that would drive the schedule. With this approach it was recognized that the risk would be high and that schedule will not be met when issues, difficulties, and unknowns arise since there is none to very little schedule margin, but these issues may be mitigated with additional funding and resources consistent with cost as a free variable. With this aggressive schedule approach, the number of technology options considered, as well as the number of tests conducted, was limited.
- *Relaxed*: With this development approach the schedule would be stretched if funding was not available when needed and thus the first launch would slip to a launch opportunity that was beyond the nominal schedule. This will most likely cause cost to go up in total, but would potentially lower the annual cost requirement. The stretched schedule would allow for additional time to address high risk items early (unknowns) and reduce schedule risk issues.

For each major development item, subject matter experts were solicited to identify key activities that would be associated with the different schedule development risk postures discussed previously. Each schedule that was developed included the necessary technology development, ground and flight demonstrations necessary to reduce risk, and flight system development. Figure 13-42 provides a notional development schedule resulting from this exercise. From a campaign and overall architecture perspective other key schedule assumptions were applied to ensure consistency between development items.

Hardware Delivery to the Launch Processing Facilities: Current and previously conceived human Mars mission concepts require significant mass to be delivered to Earth orbit prior to initiation; even with incorporation of multiple mass reducing technologies. This large mass requires multiple launches be conducted prior to the opening of the departure window for Mars. To facilitate this launch process previous assessments have determined that the hardware must arrive at the launch processing facilities approximately one year before launch and approximately six month of launch margin must be included to account for unanticipated process and operational issues which may arise. In addition, 30 days were allotted for the Earth departure window. All of these processing and operational considerations mean that, depending on the number of launches required, hardware must start arriving at the Kennedy Space Center up to two years before the departure date.

System Demonstrations: Before the first human crew would ever depart on a Mars mission, new technologies and capabilities would be developed that would enhance crew health and safety, provide capabilities for these crews to live and work that were not previously available, improve the performance of vehicles already being used, and give access to mission information of unprecedented breadth and quality. Since missions to Mars represent significant challenges to humans in terms of both the distance and time away from Earth, proper testing and validation of those new systems and technologies is required. As can be seen in Figure 13-42, the timing of the validation test of a system can have a profound effect on the resulting risk posture. In order to reduce the risk of incorporation of immature technologies or capabilities it is desired to conduct validation tests as early in the design process as possible, typically before the Preliminary Design Review. For the MPPG schedule assessment, tests which could be conducted prior to the system level PDR were considered to represent a "lower" risk posture, whereas tests which would occur after PDR were considered a "higher" posture.

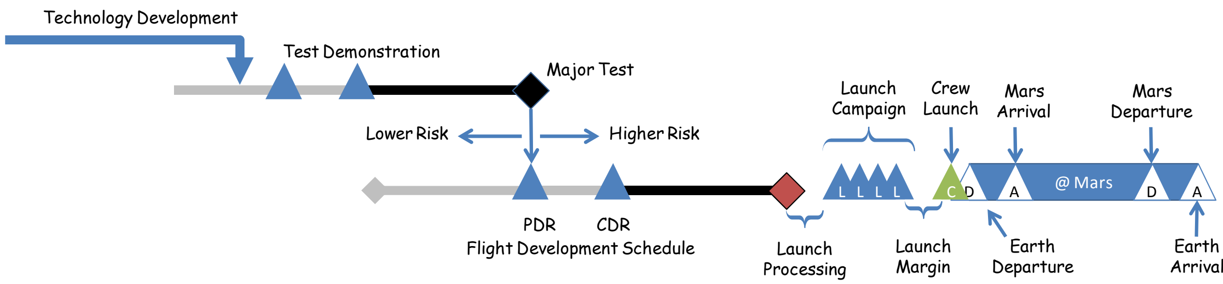

Figure 13-42 An example development schedule flow and timing.

Notional development schedules for various hardware elements associated with the major human exploration of Mars architectural components were obtained from subject matter experts (Table 13-26). These schedules included both the key transportation and human support options necessary to transport humans to and from Mars orbit (orbital missions), as well as the long lead items associated with surface missions including entry, descent and landing, advanced propulsion, ISRU, and surface power. Through this process a total of 39 schedule options were generated for the three flavors of programmatic risk posture: nominal, aggressive, and relaxed. These schedules were then integrated into logical schedule options as potential strategies for future human exploration of Mars. The intent of this exercise was not to define the definitive schedule, but rather to find key schedule logic, timing, and strategies. Figure 13-43 provides an example integrated schedule, or roadmap, for future human exploration of Mars. Examination of these notional roadmaps provided some interesting insight into the relationships between the major development and operational activities.

Near-Earth Risk Reduction Opportunities: There was a common finding from many of the schedule strategies associated with the near-Earth testing opportunities for reducing many of the risks associated with the support of humans for long-durations in deep space. It was determined that the ISS as well as missions in cis-lunar space, such as in lunar orbit or Earth-moon libration points, can play an important role in the demonstration and testing of gravity-sensitive phenomena such as crew physiology, gas/liquid separation, and large scale structure deployments. Since these mission would be by definition “near-Earth” they allow for quick and safe return of the crew should something go wrong with the operational test. These missions in LEO and Near-Earth can be used to simulate flight environments for the transit (zero-g) mission phases. In fact, each ISS crew rotation mission is essentially testing a flight to Mars from an operational and human physiology perspective. Existing platforms, such as the ISS, can provide an ideal venue for long-duration system testing including crew interaction with hardware, software, and operational procedures. Extending the testing venue beyond LEO to cis-lunar space allows for long-term exposure of systems to the deep-space environment, including radiation and zero-g. Tests in LEO and in near-Earth space can be gradually increased for short to long durations which would provide better understanding of long-duration system performance in “flight like” conditions. As tests are extended to the surface of the Moon for instance, various technologies and operational strategies associated with surface missions can also be demonstrated and validated in a planetary environment.

Human Mars Systems Tests: Each of the schedule development teams identified opportunities for large scale system and technology development tests which are necessary prior to initiation of the full-scale flight system design and development. Depending on the aggressiveness of the schedules assumed, these full-scale tests generally occurred prior to the Critical Design Review of the flight systems. When the development time, testing, launch process, and launch campaign are considered, it was found that, in general, these full-scale tests should occur approximately six years prior to delivery to KSC. It should be noted that this timing is a general finding and the specifics of the actual testing will depend on the technology chosen, risk posture adopted, and actual funding available.

Mission Mode Decision: The mission strategy adopted for the Mars Design Reference 5.0 architecture would utilize conjunction-class long-stay missions with the incorporation of advanced transportation to help reduce overall mission mass. But with the incorporation of potential missions to Mars orbit, the risks associated with human performance in this all-deep-space environment become exacerbated, and thus for these missions more emphasis would be placed on trying to minimize the exposure of humans to the deep-space environment by utilizing short-stay opposition-class missions with advanced propulsion technology as a mandatory element of that strategy. This choice of overall mission strategy, orbit first versus surface, is an important decision, which should be made in the next few years as depicted in Figure 13-43.

Sub-scale Technology Demonstrations: A major component of the MPPG exercise was to find areas of collaboration between the robotic and human exploration endeavors. Through this exercise it was determined that there is some synergy between the needs of humans to Mars orbit such as atmospheric characterization to support potential aerobraking at Mars, radiation and orbital debris characterization, but greater collaboration opportunities exist with surface missions. Some of the greatest challenges for human exploration of Mars surface relate to access of the planetary surface namely entry, descent and landing and subsequent ascent to orbit. Previous human exploration architecture assessments have shown that the EDL techniques currently being utilized by the robotic systems, namely the Mars Science Laboratory, will be inadequate for future human missions. In fact, these strategies are in dead-ended and thus new EDL technologies and techniques are required. As a key risk reduction technique for future human missions it was determined that sub-scale demonstrations of relevant EDL technologies are required. There are many options still being pursued, but it was found that this area provides a key opportunity to merge the needs of future human missions with increased landed capabilities for advanced robotic missions. Likewise, demonstration of ISRU concepts and technologies is another key area where robotic missions can play a key role in mitigating the risks to future human missions. It was found that given the overall assumption of a human orbital mission in 2033 with a surface mission two opportunities later, demonstration of these key human relevant technologies, EDL, ISRU, and Mars ascent, should be demonstrated in the early 2020's as shown in Figure 13-43.

Table 13-26 Schedule development subject matter experts.

Technology or Capability	**Source for Schedule Development**
Space Launch System	MSFC
Orion Multi Purpose Crew Vehicle	JSC
Ground Operations	KSC
Human Research Program	JSC
Deep Space Habitation	JSC
Nuclear Electric Propulsion	GRC
Nuclear Thermal Propulsion	MSFC
Solar Electric Propulsion	JPL
Advanced Chemical Propulsion	MSFC
Entry, Descent and Landing	LaRC
Methane Propulsion for Descent and Ascent	JSC
In-Situ Resource Utilization	JSC
Fission Surface Power System	GRC

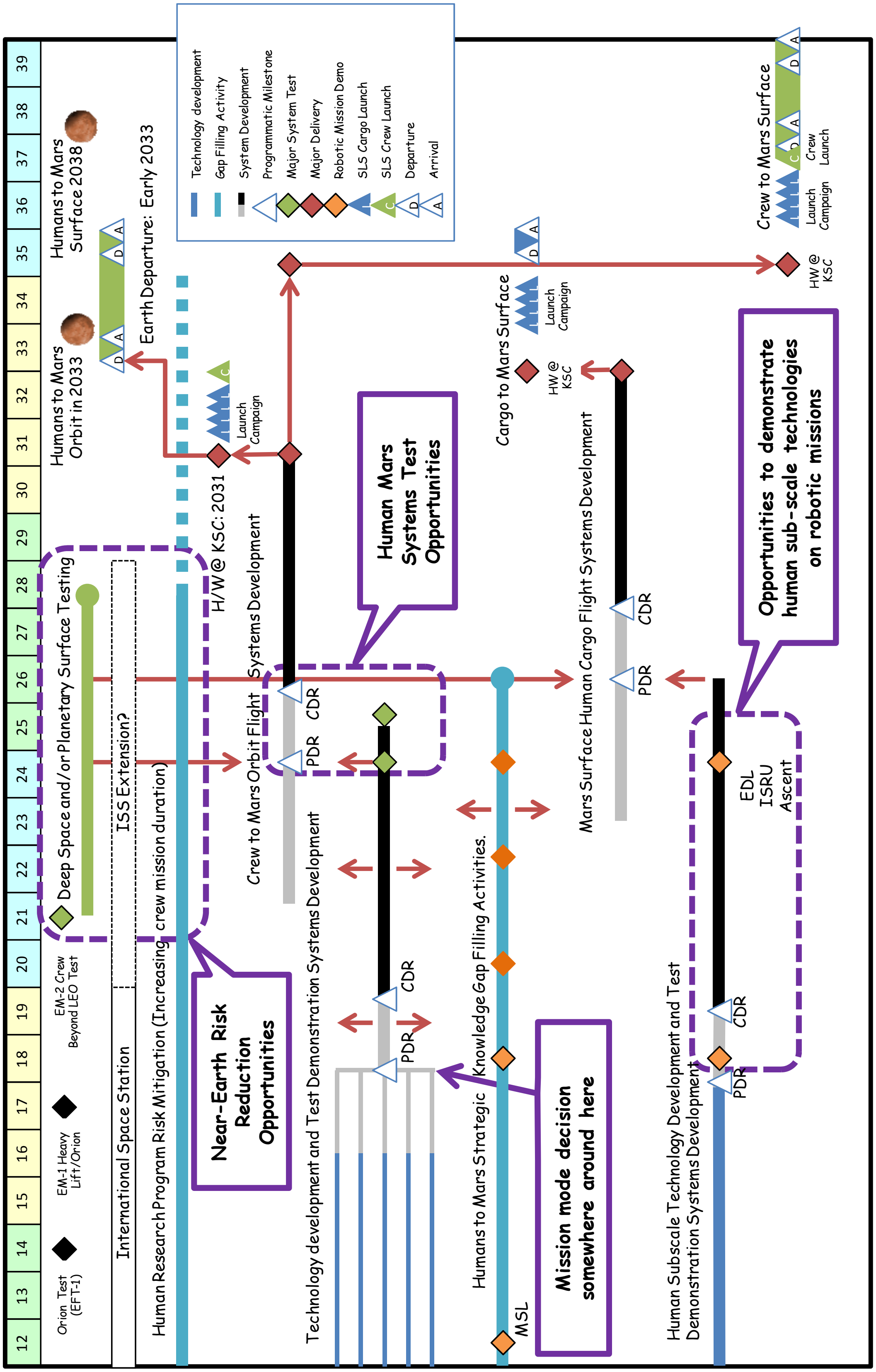

Figure 13-43 Integrated roadmap for the human exploration of Mars.

13.2.7. Precursor Investigations/Measurements necessary for Humans to Mars

Another key element of the MPPG effort was to find key areas for potential collaboration between the science and human exploration of Mars efforts. Of particular interest as the role that robotic missions can play in obtaining critical environmental data and reducing future mission risks. To address the missing information needed to send humans to Mars, NASA requested a joint SMD-HEO activity sponsored by the Mars Exploration Program Analysis Group (MEPAG) on the topic of "precursor measurements" that could both address the 2013-2022 Visions and Voyages Planetary Decadal Survey [NRC, 2011] as well as the human exploration goals. The 2012 MEPAG Precursor Science Analysis Group (P-SAG) was formed to: 1) examine the strategic knowledge gaps (SKGs) in knowledge of Mars required to support the first human missions to martian orbit (Goal IV-) and to the surface (Goal IV), 2) identify the key science objectives (using existing MEPAG and NRC scientific planning) that could be addressed in synergy with each of the potential investigations from the SKGs, and 3) identify key technology development/demonstration opportunities necessary to support humans-to-Mars objectives. Additionally, P-SAG (2012) worked to 4) classify each of the opportunities identified above by implied or potential platform (e.g. orbiter, stationary lander, rover, etc.), and to evaluate relative priority [P-SAG, 2012][16].

13.2.7.1. Strategic Knowledge Gaps and Gap Filling Activities

Strategic Knowledge Gaps (SKGs) are defined as gaps in the knowledge needed to achieve a specific goal. Working through community input, the MEPAG [MEPAG, 2010][17] has identified consensus goals and objectives for exploration of Mars as:

- Goal I: Determine if life ever arose on Mars
- Goal II: Understanding the process and history of climate on Mars
- Goal III: Determine the evolution of the surface and interior or Mars
- Goal IV: Prepare for human exploration

For the MPPG effort, further assessments associated with Goal IV were required, but in addition to preparing for missions to the surface, assessments for potential orbital missions to explore the moons of Mars as well as long-term exploration of the surface of Mars were examined. The Strategic Knowledge Gaps (SKGs) associated with each of the following goals were defined by the P-SAG: 1) First human mission to martian orbit (Goal IV-); 2) First human mission to land on either Phobos or Deimos; 3) First human mission to the martian surface (Goal IV); and 4) Sustained human presence on Mars (Goal IV+). The SKGs were broken down into Gap-Filling Activities (GFAs), and each was evaluated for priority, required timing, and platform. The relationship of the above to the science objectives for the martian system (using existing MEPAG, SBAG, and NRC scientific planning), was evaluated, and five areas of significant overlap between HEO and science objectives were identified. 1) Mars: Seeking the signs of past life; 2) Mars: Seeking the signs of present life; 3) Mars: Atmospheric dynamics, weather, dust climatology; 4) Mars: Surface geology/chemistry; and 5) Phobos/Deimos: General exploration of Phobos/Deimos. Within these areas it would be possible to develop exciting mission concepts with dual purpose. The priorities relating to the Mars flight program have been organized by mission type, as an aid to future mission planners: orbiter, lander/rover, Mars Sample Return (MSR), and Phobos/Deimos.

The high-priority gaps for a human mission to Mars orbit relate to a) atmospheric data and models for evaluation of aerocapture, and b) technology demonstrations for optical communication and in situ resource utilization, as well as orbital rendezvous, ascent demonstration, dust mitigation, radiation exposure, and sample handling. A human mission to the Phobos/Deimos surface would require a precursor mission that would land on one or both moons. The early robotic precursor program needed to support a human mission to the martian surface would consist of at least one orbiter, a surface sample return (the first mission element of which would need to be a sample-caching rover), a lander/rover-based in situ set of measurements (which could be made from the sample-caching rover), and certain technology demonstrations. For several of the SKGs, simultaneous observations from orbit and the martian surface need to be made. This requires multi-mission planning. (See PSAG, 2012; slides 4-12, http://mepag.jpl.nasa.gov)

13.2.7.1.1. Dust Toxicity to Humans

We do not understand in sufficient detail the factors affecting crew health and performance, specifically including the biological effects of the potentially toxic properties of the martian dust. The jagged nature of lunar dust particules has been shown to have a significant effect on crew health. Key in understanding whether similar concerns exist for martian particulates is gaining knowledge of the surface particulate material properties, including electrical, chemical and physical characteristics. (See PSAG 2012 Appendix 2, slides 26, 30, http://mepag.jpl.nasa.gov)

13.2.7.1.2. Radiation Environment on the Surface of Mars

Because Mars has no planetary magnetic field, unlike Earth, any radiation attenuation that would exist would be due to the atmosphere and from the planet itself. The Mars atmosphere density is 16 g/cm^2 (versus 1000 g/cm^2 on Earth) but varies widely from season to season. Radiation transport depends on both the altitude and atmospheric density; radiation is either absorbed, fragments to produce secondary particles, or propagates to the surface which would also result in secondary particles. Mars Science Laboratory has already made significant progress on this SKG through the he Radiation Assessment Detector (RAD) instrument on board the Curiosity rover. RAD's primary science objectives are to characterize the energetic particle spectrum at the surface of Mars, determine the radiation dose for humans on the surface of Mars, and enable validation of Mars atmospheric transmission models and radiation transport codes. Preliminary data returned from the Martian surface shows that at the season and atmospheric conditions Curiosity landed, the martian atmosphere attenuates 5 units of radiation in addition to the radiation attenuated from the planetary body (Figure 13-44). Whether this value can be extrapolated to other times and conditions of the martian atmosphere remains to be understood. (See PSAG 2012, Appendix 2, slides 26-29, http://mepag.jpl.nasa.gov, and http://mslrad.boulder.swri.edu/)

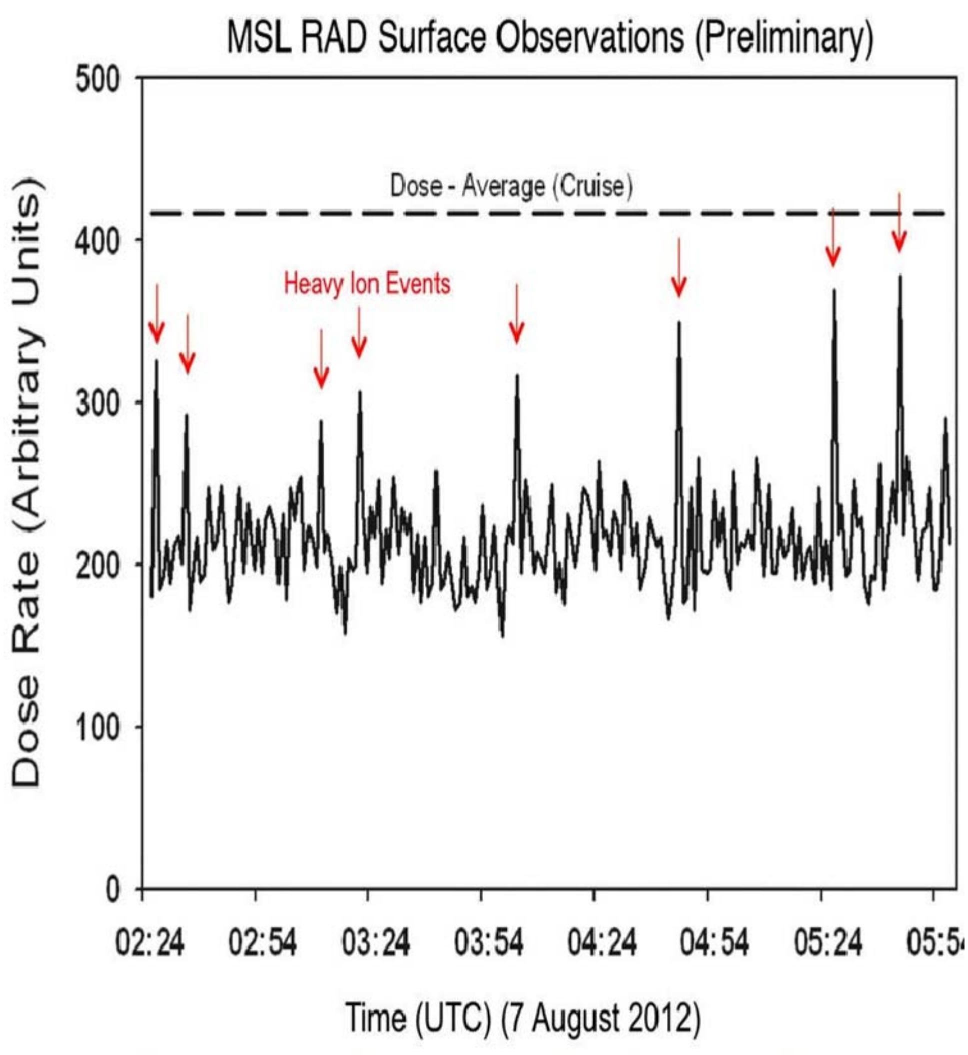

Figure 13-44 Preliminary Mars Science Laboratory radiation observations.

13.2.7.1.3. Landing site specifics, weather prediction

Whether measurements that have been acquired at disparate locations around Mars can be extrapolated to any specific potential landing site for human exploration remains an open question. In addition, while on-going and future atmospheric measurements are an excellent record of climate, the particular weather on the day of a potential human entry into the martian environment is unknown. Just as on Earth, validating and collecting data for predictive climate and weather models is the most accurate method to understand the specific conditions that might exist on Mars in the future. Validation allows the models to be confidently used to create the extreme conditions (>99% distribution tail) necessary to select and design EDL, aerocapture, aerobraking, and launch systems. Atmospheric model validation and data assimilation can take several years following data acquisition, and should be factored into mission phasing. Long-lived orbiters with global diurnal coverage would provide the largest volume of atmospheric data to support model development and validation. Also, multiple landers providing simultaneous measurements with the orbiters would be needed to acquire near-surface data correlated with upper atmosphere measurements, for model development and validation. See PSAG (2012), Appendix 2, slides 2-8, http://mepag.jpl.nasa.gov)

13.2.7.2. Sample return on Critical Path to Human presence at the surface of Mars

To prepare for a human mission to the martian surface, based on what is known today sample return is the only implementation that would address the required back planetary protection (PP) SKG. That is, we do not know whether the Martian environments to be contacted by humans would be free, to within acceptable risk standards, of biohazards that might have adverse effects on some aspect of the Earth's biosphere if uncontained Martian material were returned to Earth. Despite the best intentions and best engineering, it is likely that some uncontained martian dust and regolith would be returned to Earth with the crew. The safest and possible the only acceptable way to ensure non-release of biohazardous material into the Earth's biosphere is to return carefully contained samples to Earth prior to the first human mission. Prior to that mission, one or more diverse sets of regolith, rock and dust samples should be collected from sites representative of the diversity anticipated at human landing sites and returned to Earth for comprehensive biohazard testing similar to that outlined in Rummel et al., (2002) to determine whether any indigenous life is present and, if so, whether it presents a hazard to the Earth's biosphere. A significantly more risky, and possibly unacceptable, approach to lessen the risk of returning uncontained living and potentially hazardous organisms with the crew is to identify zones of minimum biological risk (ZMBR's) as potential landing sites. Biologic risk to be identified by orbital measurements for signs of recent water activity, orbital land lander measurements for presence of ground ice, and lander measurements following some to-be-determined life detection protocols (total carbon, isotopes, etc) on near surface materials at potential landing sites. (See PSAG 2012, Appendix 2, slides 21, 62, http://mepag.jpl.nasa.gov)

13.2.7.3. Future astronaut surface science objectives: Changes since 2007

The Human Exploration of Mars – Science Analysis Group [HEM-SAG, 2008][18] prepared a preliminary analysis of the science that could be done by a human exploration crew on the martian surface. Since then, the following significant things have changed:

- The NRC has re-evaluated science priorities and strategies for the coming decade (2013-2022)
- MSL has successfully landed, and has begun its scientific mission
- The InSight mission has been selected by the Discovery Program, and if successful, it would result in the delivery of single station Mars seismic data and a one-site heat flow measurement.
- MPPG has prepared an analysis of possible transitions from a science-driven Mars program to a human precursor-driven Mars program.

Implications:

- Astrobiology and the search for ancient life remains the top strategic driver for the scientific exploration of Mars.
- MSR continues to dominate scientific thinking for the next decade, although whether this mission or set of missions is acceptable politically is a question. It will therefore be important for planning astronaut-implemented science to continue to think through the possible discoveries of MSR, and how they might

best be followed up.

- MSL's most important early discovery is re-confirmation that Mars had flowing liquid water early in its history. This will reinforce the ancient life investigation pathway. More significant discoveries (e.g. related to organic molecules) are expected within approximately the next 6-12 months.
- HEM-SAG called for astronaut deployment of long-lived seismic investigations. InSight, if it returns data successfully, will cause us to need to re-think that.
- The transition from a science-driven program to a human prep program are intrinsically caught up in the timing/planning for MSR. Again, the key to planning astronaut-tended surface science

13.2.8. Human and Robotic Mission Collaboration Opportunities (Baker)

13.2.8.1. Beyond LEO Ops (Baker)

13.2.8.2. Strategic Gap Filling Activities (Baker)

13.2.8.3. Technology Demonstrations (Baker)

13.2.8.4. SLS Secondary Payloads (Baker)

13.3. Risk Assessments

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All Mars risk analysis effects in support of the Human Spaceflight Architecture Team (HAT) were halted in October of 2011. Early in 2012, Human Exploration Development Support (HEDS) management, in attempting to formulate various strategies, defined the need for an integrated risk picture, first starting with the identification of key risk drivers for human Mars missions. Risk analysis experts were contacted and discussions began on how to determine the integrated risk picture. This effort led to a plan to determine the overall integrated risk picture with the emphasis of this effort is on risk identification, prioritization, and developing mitigation strategies.

Previous efforts during the Mars architecture study (documented in Human Exploration of Mars Design Reference Architecture 5.0) included a two phase risk modeling approach. An initial, high level analysis was conducted which help identify key risk drivers, followed by a refinement of the risk critical models. The first phase was scenario based risk assessment approach was performed by experienced analysts to identify risk drivers for the proposed Mars missions. The second phase of the analysis focused on risk mitigation strategies based on trade trees and elimination of options that do not meet risk, cost, or performance specifications. Estimates of probability of Loss of mission and Loss of Crew values were produced, but no clear cut decisions that could be made from a risk standpoint were determined.

From the results of this initial effort, it was determined, that current design philosophies and technologies would not provide an acceptable level of reliability for a Mars mission. The risk analysis also highlighted many areas that would benefit from additional, more detailed analyses and that continued refinement of the risk driver calculations can be made as system details become more comprehensive.

This current risk assessment effort (started in FY'12) is an analysis effort aimed at emphasizing identification of risk drivers and their prioritization. After the risk analysis model is developed and running, further definition of the elements will be undertaken as well as discussions with subject matter experts to get their buy-in on the representative subsystems and assigned failure rate predictions. When the element data is refreshed with this new information, the model will be re-run to provide a higher fidelity indication of risk drivers. Ultimately, the model will enable different options and mission architectures to be assessed for risk, and enable assessment of various risk mitigation strategies and help determine precursor mission synergies.

13.3.1. Methodology

In order to provide a risk analysis with a capability to rapidly support high level architecture and strategy discussions, the elements will be modeled in the MS Excel-based Rapid Response Tool. The Rapid Response Tool (RRT), is a risk assessment tool developed by JSC Safety & Mission Assurance organization, to rapidly respond to requests to estimate mission risk and vehicle reliability on trade studies during early phases of project design and development. The RRT has been used for the Morpheus Project, the Lunar Orbit Rendezvous Design Reference Mission (DRM) for the Constellation Altair Lunar Lander, and the EMSO Deep Space Habitat Near-Earth Asteroid DRM.

The systems and components in the RRT are defined on individual system worksheets. Individual mission event model parameters and selected subsystems in operation during specific mission events states are defined. The mission events and systems are linked to an EC Tree worksheet which in turn calculates the end state probabilities such as loss of mission and loss of crew. The risk associated with subsystems, as well as the contribution of mission events/phase to overall element risk, are all available.

The results from the RRT element models (end state probabilities [loss of mission, loss of crew] for a specific element, for a specific mission event phase) are combined with a high level event tree in a Systems Analysis Programs for Hands-On Integrated Reliability Evaluations (SAPHIRE) Probabilistic Risk Assessment tool to identify risk drivers (by element , and by particular phase). This determines which particular element for which phase is the driving risk. To determine what system is the driver for a risk driving element; the RRT element model is utilized.

Because of the use of the RRT to analyze the more detailed aspects of each element, system, and subsystem, the model is highly modular and flexible. Changes in systems, subsystems or components can be quickly reflected in the risk model, allowing for "near real time" risk trades.

In initial development of the model, utilizing point estimates for failure rates, random failures of the hardware were the risk drivers. The initial runs are based on the assumption that the hardware is mature (having been flown and operated in an environment similar to what it would be exposed to for the Mars mission) and random failures of the hardware drive the risk. As the model is developed, maturation of the hardware will have to be taken into account. For example, Aero-assist entry into the Mars atmosphere has not been done before and little or no testing has occurred. The technology is very immature as are estimates of the risk. Liquid Oxygen/Methane fueled engines are new technology, as are Nuclear Thermal Rockets. Testing these engines in space or on Mars has not occurred. Further development of the risk analysis model will take into account that these technologies are immature and will mature as they are tested and used in-space.

13.3.2. Risk Analysis

13.3.2.1. Developing event timelines

Starting with the Design Reference Architecture (DRA) 5.0 reference mission, the timeline for a crewed portion of the mission to Mars was modeled. In the initial model, the uncrewed portions of the mission, including the delivery of the Mars Ascent Vehicle (MAV), the In-Situ resource Utilization (ISRU), the first Surface Fission Power unit, as well as their launchers and in-space transportation stages are not modeled. The assumption was made that these elements were successfully transported to and landed on the Martian surface, and the Surface Fission power unit set up and producing power for the ISRU to generate sufficient oxidizer propellant for the MAV, and the MAV was checked out and ready to support a crew departure when required.

The launch campaign (for the crewed portion of the mission) was modeled to take into account assumptions for the ground architecture (number or launch pads and number of vertical assembly building high bays configured to support these mission elements) as well as workforce assumptions (number of shifts and length of work week). Realistic time margin was added to the element and launcher processing time to ensure a historic reasonable probability of launching within the launch window. All this was taken into account to determine realistic launch spacing for the various elements. This campaign modeling, in-turn, helped determine the accumulated time in Earth orbit for the various elements. Operating time for all of the elements was calculated from launch through disposal of the respective element.

For the initial model, a Space Launch System (SLS) class launcher was assumed. Other launchers will modeled under future work to include their impact on ground architecture, work force, processing, and most importantly, number of launches to put an equivalent payload mass in low Earth orbit.
Per the timeline, once the crew was launched, the counter was started for the accumulated crew time during the

various phases of the mission. Significant events such as major propulsion system firings [Trans Mars Injection (TMI), Mars Orbit Insertion (MOI)]; rendezvous; Mars Entry, Descent and Landing (EDL); ascent from the Martian surface were all captured in the timeline.

Event timelines were developed for the Conjunction and Opposition class missions, as well as utilizing both Chemical and Nuclear Thermal Propulsion for in-space transportation.

13.3.2.2. Modeling elements

For the initial version of the model, the elements include the Multi-Purpose Crew Vehicle (MPCV), the Mars Ascent Vehicle (MAV), the Transit Habitat, the Mars Descent Lander/Habitat, the Nuclear Thermal Propulsion Stage, the In-line Tank, Saddle Truss/Drop Tank, and Re-boost module, Docking Module, and the Cryo Propulsion Stage (CPS). With the exception of the MPCV, which is currently being extensively modeled in support of the MPCV program, most of the rest of the elements are at a low maturity level. Increasing the maturity required that elements be developed from descriptions in the DRA 5.0 reference mission as well as utilizing experience on other former Constellation elements. Element level models down to the subsystem level are required for predicting element failure rates during specific phases of the mission. More mature models of the elements will occur as their definition becomes more detailed.

The elements are modeled in the MS Excel based Rapid Response Tool where the components making up the systems are defined as well as their failure rates predicted, based on historical data for similar type components in similar environments. Each mission phase is defined as to what systems are operating (on or off) as well as the phase duration (in hours)

13.3.2.3. Data sources

Data for the system components is taken from a variety of sources and is chosen on a best fit basis, based on component similarity and environment. Source of the reliability data failure rates, used in the Rapid Response Tool element models, comes from the International Space Station Program, the Space Shuttle Program, the Constellation Program (which in turn was derived from the International Space Station and Space Shuttle Programs), other space systems such as satellites or launch vehicles, and other related analysis (crew medical, radiation, Micro-Meteoroid Orbital Debris, etc.). Currently, point estimate values are used and eventually, data uncertainty may be introduced.
13.4.3.4 Other Risk

Other items such as human reliability, software failures and Micro-Meteoroid/ Orbital Debris and crew medical risk were also included in the model.

Human error was determined by analysis of historic data to be about 10% of the Habitat total mission risk. Software failures were also determined by historic analysis to be 5% of the Habitat total mission risk. An increased reliance on software to operate spacecraft systems results in a higher percentage of software failures and lower percentage of human error, but still resulting in about a 15% of the Habitat total mission risk. Conversely, decreasing the reliance on software by requiring more human operation resulted in a lower percentage of software failures and a corresponding increase in human error.

Micro-meteoroid/Orbital debris rates were estimated from Lunar Sortie Loss of Crew/Loss of Mission achievability analysis daily exposure rate for Low Earth Orbit (LEO) (for the LEO loiter phase of the mission), Cis-Lunar and Beyond Cis-Lunar mission (for the Mars transit, Mars capture, and Mars orbit phases of the mission) and both rates scaled for the Mars DRM and MTV stack surface area.

Crew medical risk (for the mission period, about 900 days) is determined from the Integrated Medical Model (IMM), June 2011 scaled for the Mars DRM crew size and time of exposure.

13.3.2.4. Unknowns

With very little data on Aero-assist entry and supersonic retro-grade propulsive engine start/operation, the impact of these events has been estimated. As more testing, analysis, and element design occurs, these estimates can be revisited. Impacts to crew health (such as Visual Impairment, Inter-cranial Pressure (VIIP)) such as sustained exposure to weightless environment are also unknown at this time.

13.3.2.5. Initial findings

13.3.2.5.1. Crew Sample Return mission

In order to support an urgent need of the Mars Program and Planning Group (MPPG), an initial assessment, required shortly after this effort was started, was performed to model the risk drivers for loss of crew on a Mars Sample return mission where a human crew was to travel to Mars, retrieve a previously collected Mars surface sample in low orbit around Mars. Utilizing the DRA 5.0 reference architecture, the crewed portion of the conjunction class, Nuclear Thermal Rocket propelled Mars mission was utilized as a starting point, with the primary difference being (for this initial assessment), the crew remained in Mars orbit for the duration of their Mars stay time (500 days) as opposed to descending, landing and staying on the surface. For this assessment, only the elements involved with the crew portion of the mission were modeled, i.e., the Nuclear Thermal rocket (NTR) for the crew transit, the inline propulsion tank, the saddle truss and drop tank, the Mars transit vehicle (MTV), a Low Earth Orbit re-boost vehicle, the return crew exploration vehicle (CEV), and finally, the crew delivery crew exploration vehicle. A descent lander, aero-shell, surface habitat, and Mars Ascent Vehicle were not modeled as this sample return mission does not utilize them. The Mars sample collection portion of the mission was not modeled. The collected sample was assumed to be in low Mars orbit and the crew would conduct a rendezvous and proximity operation and retrieve the sample.

To enable rapid modeling (in time to support the requested need) of the mission, several ground rules and assumptions were used, including three strings for each system, certain systems that would not result in risks leading to loss of crew were not included in the model (such as Communications and Tracking, general instrumentation and some heaters), no cross strapping across elements to perform critical functions, however cross strapping within a system or element was used, no spares or repairs were included. For this model, all components were operated at 100% duty cycle when on, radiation and thermal protection were assumed to be included in the structure failure rate, and a crew size of four crew members. Aborts (other than upon crew launch from Earth) were not modeled, and Mars Transfer Vehicle capture in Mars orbit was accomplished.

Launch availability was modeled with a launch campaign analysis utilizing the following ground rules: the duration of the Mars injection window is 60 days; the crew can be launched no earlier than 30-days prior to the opening of the Mars injection window; Two days are required from crew launch prior to performing the Trans Mars Injection burn; there was no constraint on how early we can launch cargo missions. Three working shifts per day and a five day work week schedule was assumed for the work force.

13.3.2.5.2. Results

The Mars Transfer Vehicle (MTV) Habitat failing in Mars orbit was the leading risk driver for loss of crew, as the habitat was the most complicated element and was operating for the longest time period (500 days). Other risk drivers were the NTR fails in Mars orbit, the Crew return MPCV fails in Mars Orbit, and docking module fails in Mars orbit. All of these show the impact of the longest mission phase (500 days) and its impact on determining risk drivers. The impacts on crew health for being subjected to prolong weightless (~900 days) during the mission are included in the IMM so they have not been modeled.

13.3.2.6. Crew mission to the surface

To help illustrate/compare the risk of a sending a crew on a sample return mission to Mars, and a mission were the crew is sent to the surface of Mars, a crew mission to the surface was modeled. Additional elements required for sending the crew to the surface as well as support them while on the surface and returning the crew to the Mars Transfer vehicle was added to the model. These additional elements (Descent/Lander Hab, surface power source, Mars Ascent Vehicle) were modeled and added to the model. Aborts from the Mars surface (if the Descent /Lander Hab or surface power source were to fail) were included in the model since DRA 5.0 architecture contained adequate consumables in the MTV to support the crew in the event of a shortened surface stay.

13.3.2.6.1. Results

Risk drivers for the crew to surface mission included the Mars Transfer Vehicle (MTV) Habitat failing in Mars orbit was the leading risk driver for loss of crew, as the habitat was the most complicated element and was operating for the longest time period (500 days). The MTV habitat failing during the transit to Mars was the next risk driver, owing to complexity and mission phase length. Factors such as allowing a surface abort from the surface reduced the impact of failures on the surface Habitat and surface power source.

13.3.2.7. Potential low-hanging fruit

If the impact maintenance upon the failure rate was modeled, assuming the spare parts were available, and the crew had the necessary tools and skills to perform the repairs, the risk of Loss of Crew or Mission can be reduced (improved). Rough estimates of the maintenance level (percentage of hardware and components that are accessible and repairable with available spares, tools and crew skill sets) show a probable decrease in risk with increasing levels of repairable hardware. In a high level analysis, assuming International Space Station levels of reparability (roughly 65%), the three longest mission phases (transit to Mars, Mars orbit [or surface stay], and transit to Earth) show decreases in Loss of crew risk of over 3.5 times for the Hab module. As both the elements and model matures, impacts of introducing options such as maintainability, upon the overall mission risk will be modeled to show how best to apply a maintenance concept to reduce risk.

13.3.3. Future risk analysis work

Future work will include further development/refinement of the model and elements, to refine the initial risk estimates. Maturation of the hardware elements will also be included in the model. Model refinement should allow rapid risk assessment of various options. Close coordination of the elements with subject matter experts will improve the level of detail and should improve the accuracy of the risk estimation.

As risk drivers are identified, they can be further analyzed to determine the cause of the risk and methods to potentially mitigate the risk can be assessed. Operating times for hardware, redundancy levels, mission phase lengths, architecture changes will all be modeled to assess their impacts upon risk. The model can rapidly be re-run with changes to the elements to examine the impacts to the risk. When the risk drivers have been determined for a specific mission and architecture, those risks can be analyzed to assess how the various precursor missions could help mitigate this risk. Depending on the nature of the risk and the details of the particular precursor mission, some precursor missions could help mitigate more risk than other missions. The risk assessment model will help the user to determine which precursor strategies or combination of missions offer the best risk mitigation.

With several different precursor activities available (International Space Station (ISS), Multi-Purpose Crew Vehicle (MPCV) Cis-Lunar flight, Waypoint (L-2), Near Earth Objects (NEO), Lunar Missions, Crew missions to the Mars moons Phobos and Deimos, and Robotic Missions to Mars), many opportunities are present to mitigate or buy down risk. The risk assessment will aid in showing where the best return of investment can be achieved. Maturation of the hardware will result from utilizing the hardware in environments similar to where it will be used on the Mars mission without exposing it to all the risks of the Mars mission. Long duration in-space exposure can come from locating (and operating) the hardware on the ISS, at the Waypoint location, on Near Earth Object missions, on Lunar missions. Entry systems/techniques can be demonstrated in the high Earth atmosphere, as well as on robotic missions to Mars (where the descent and landing portions of the mission can be demonstrated as well). Waypoint and NEO missions can demonstrate long duration space missions in an environment similar to the Mars transit and orbit phases. Lunar missions provide the opportunity to operate surface power systems and potential landing systems. Missions to the Mars moons can demonstrate/exercise all of the hardware required to perform a Mars mission except the atmospheric entry portion. Robotic missions to Mars (in addition to demonstrating Entry, Descent and Landing systems) can demonstrate/exercise surface power systems and in-situ resource utilization systems which would help reduce risk.

Reliability growth will come from exercising the components and systems in the environment that they are designed to be used. Testing can accomplish some of this but with some of the components and environments on Earth, it is difficult to demonstrate a 100% totally accurate environment (Such as the impact of gravity. Mars has one-third of the gravity of Earth). Reliability growth will result from subjecting flight or flight-like hardware and software to the "test, analyze, fix" cycle that has traditionally been applied to assure design suitability and robustness. Exercising of the hardware through precursor missions, exposing the hardware to mission operating cycles and environments, will contribute to reliability growth of the hardware by exposing weaknesses and the subsequent redesign of the failed components.

13.4. Planetary Protection Requirements for Human and Robotic Missions to Mars

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13.4.1. Overview:

Ensuring the scientific integrity of Mars exploration and protecting the Earth and the human population from potential biohazards requires the incorporation of planetary protection (PP) into spaceflight missions, both robotic and human. All missions going to or returning from Mars are required to comply with stringent planetary protection requirements that are based on provisions of the Outer Space Treaty. Thus, planetary protection is an essential element in the architecture of future Mars exploration—both robotic and human—and must be incorporated from the earliest phases of mission planning. To meet these requirements, any Design Review Architecture (DRA) for human Mars missions must incorporate considerations of both forward and back contamination controls in multiple mission phases, in order to 1) protect the astronaut crew, as well as the biosphere of Earth upon the return of the crew and scientific samples; 2) monitor and assess astronaut health throughout the mission; and 3) enable the success of scientific investigations and sampling focused on habitability and detection of potential martian life. Based on latest international policies and NASA directives, as well as ongoing implementation of both human and robotic missions, and recognized needs for R & TD, planetary protection information is essential input for future human exploration of Mars. Planetary Protection provisions are not merely recommendations or suggestions, but rather mandatory planning elements that should be considered in all human mission systems and subsystem development activities from the start. They are cross-cutting in nature, contribute to requirements generations, have feed forward implications, and represent significant time and funding considerations for any future design reference architecture.

13.4.2. Background

Since early in the space age, the Committee on Space Research (COSPAR) of the International Council for Science (ICSU) has maintained a consensus planetary protection policy for joint reference and implementation [COSPAR, 2011][19]. COSPAR's policy stipulates the need to control forward contamination (life or organic contamination carried from Earth) that might invalidate current or future scientific exploration on a particular solar system body, or might disrupt planetary environments or potential endogenous (alien) ecosystems. In addition, concerns about backward contamination (extraterrestrial life carried back to Earth) focus on avoiding the potential for harmful contamination of the Earth's biosphere. For human missions, this also includes the possible immediate and long-term effects of biologically-active materials encountered during exploration on the health of astronaut explorers.

For nearly five decades, NASA's robotic missions have complied with international planetary protection specifications and controls while exploring the Moon and other celestial bodies [NASA, 2012][20]. Moreover, multiple National Research Council (NRC) studies of the exploration of Mars and other bodies of the solar system have reiterated the importance of taking a conservative approach to planetary protection implementation for both forward and backward contamination controls [NRC, 1992, 1997, 2002a 2002b, 2007, 2008, 2009][21,22,23,24,25,26,27]. Nonetheless, few human missions have been subjected to PP requirements. The crews of Apollo 11, 12, and 14 were quarantined upon their return from the Moon, after the first human landings on another celestial body. After extensive analyses of astronauts and samples demonstrated that lunar materials posed no biological threat to mission personnel or Earth, strict PP requirements were eliminated for lunar missions. Because post-Apollo human missions have only journeyed as far as low-Earth orbit (LEO), the human spaceflight program has no recent experience with PP implementation (although some of the same principles govern the Shuttle-era health stabilization program).

Human missions involving the ISS, Shuttle, or other platforms in Earth orbit are not constrained by planetary protection controls. It is noteworthy that the science, technology and legal considerations for planetary protection during future long duration human missions—especially for Mars—are significantly different than those used during the Apollo program. Thus, it will be particularly important to initiate discussions/interactions with varied engineering and technical communities as well as space medicine, biomedical, operations and human/factors communities—to ensure that all communities include up-to-date information on implementation of planetary protection for future exploration beyond LEO. Moreover, since it is recommended that a crewmember be assigned the responsibilities for planetary protection oversight on long duration missions, there will be implications for team training, autonomy, and related operational considerations.

The primary goals of the COSPAR planetary protection policy do not change when humans are involved. If human explorers are to be beneficial to the understanding of planetary environments and potential life, or to ensure their own safety while conducting planetary exploration, then consideration of planetary protection is essential. In doing so, the unavoidable, and mostly beneficial association of humans with a huge diversity of commensal microbes means that special implementation controls will have to be developed for human exploration missions— particularly for future long-duration missions to Mars. There is a need to acknowledge and emphasize important cross-cutting, feed-forward considerations that planetary protection concerns will involve. To mitigate the potential for danger to astronauts and to Earth, as well as to avoid forward contamination of other bodies, planetary protection must be acknowledged as an important element for the success of human missions—and evaluation of planetary protection requirements should be considered critical in all human mission systems and subsystem development activities from the start.

13.4.3. COSPAR Planetary Protection Policy for Robotic and Human Missions

As indicated in NASA Policy Directive (NPD) 8020.7G (section 5c) [NASA, 1999][28], ensuring compliance with the Outer Space Treaty planetary protection is a mandatory component for all solar system exploration missions. International planetary protection policy and guidelines for compliance with Treaty obligations are maintained by the COSPAR, which also advises the United Nations on matters of space exploration. In addition to complying with applicable forward contamination control measures associated with science data collection and operations, NASA's planning of human mission architectures are expected to be compliant with approved COSPAR planetary protection principles and guidelines shown in Figure 13-45. Further elaboration of specific requirements by NASA is anticipated as architecture planning continues.

In developing preliminary guidelines for human missions to Mars, COSPAR has noted that the greater capability of human explorers to contribute to the astrobiological exploration of Mars will be realized only if human-associated contamination is controlled and understood. A robust program of planetary protection measures, including forward contamination control, medical monitoring, spatial planning for human exploration, and precautions against back contamination, has been described in NASA and ESA-led studies [Race, 2008][29], with an assumption that prior to human exploration there is a need for efforts to develop, rehearse and refine planetary protection controls. Effectively, these principles involve "defense in depth" and the continuous evaluation throughout a mission of the contamination status of the crew and the planetary surface (and sub-surface) that they will explore.

Planetary Protection Principles and Guidelines for Human Missions to Mars [from COSPAR Policy 2011—page A-5.)

The intent of this planetary protection policy is the same whether a mission to Mars is conducted robotically or with human explorers. Accordingly, planetary protection goals should not be relaxed to accommodate a human mission to Mars. Rather, they become even more directly relevant to such missions—even if specific implementation requirements must differ. General principles include:

- Safeguarding the Earth from potential back contamination is the highest planetary protection priority in Mars exploration.
- The greater capability of human explorers can contribute to the astrobiological exploration of Mars only if human-associated contamination is controlled and understood.
- For a landed mission conducting surface operations, it will not be possible for all human associated processes and mission operations to be conducted within entirely closed systems.
- Crewmembers exploring Mars, or their support systems, will inevitably be exposed to martian materials.

In accordance with these principles, specific implementation guidelines for human missions to Mars include:

- Human missions will carry microbial populations that will vary in both kind and quantity, and it will not be practicable to specify all aspects of an allowable microbial population or potential contaminants at launch. Once any baseline conditions for launch are established and met, continued monitoring and evaluation of microbes carried by human missions will be required to address both forward and backward contamination concerns.
- A quarantine capability for both the entire crew and for individual crewmembers shall be provided during and after the mission, in case potential contact with a martian life form occurs.
- A comprehensive planetary protection protocol for human missions should be developed that encompasses both forward and backward contamination concerns, and addresses the combined human and robotic aspects of the mission, including subsurface exploration, sample handling, and the return of the samples and crew to Earth.
- Neither robotic systems nor human activities should contaminate "Special Regions" on Mars, as defined by this COSPAR policy.
- Any uncharacterized martian site should be evaluated by robotic precursors prior to crew access. Information may be obtained by either precursor robotic missions or a robotic component on a human mission.
- Any pristine samples or sampling components from any uncharacterized sites or Special Regions on Mars should be treated according to current planetary protection category V, restricted Earth return, with the proper handling and testing protocols.
- An onboard crewmember should be given primary responsibility for the implementation of planetary protection provisions affecting the crew during the mission.
- Planetary protection requirements for initial human missions should be based on a conservative approach consistent with a lack of knowledge of martian environments and possible life, as well as the performance of human support systems in those environments. Planetary protection requirements for later missions should not be relaxed without scientific review, justification, and consensus.

Figure 13-45 Planetary protection principles.

13.4.4. Applying PP Considerations to Future Human Design Reference Architectures

Within the past decade, several workshops and studies have specifically analyzed hypothetical mission scenarios and reconsidered PP for long duration human missions, especially to Mars [Criswell, 2005][30], [Hogan, 2006][31], [Kminek, 2007][32] These cross-disciplinary workshops and studies noted the need to take a broad conceptual approach to PP during human missions and to develop special PP technologies and operations to address important contamination-associated factors. Ultimately, plans and designs for PP provisions on future human missions must be developed in ways that build upon integrated understanding and open exchange of information among the many technical and scientific communities involved. In practice, this means that any DRA must reflect the fact that PP requirements are not optional—they are required to ensure adherence to the Outer Space Treaty, and represent important drivers of many R & TD activities, operations and implementation schemes. Thus it will be critical that program architectures involving combined human and robotic missions benefit from crosscutting technologies and leveraged cost from the earliest phases of mission plans.

Significant investments will be needed (as supported by the NRC Decadal Survey) in multiple areas of importance that emanate from a combination of COSPAR planetary protection policies for forward contamination control and associated guidelines for human missions to Mars focusing on back contamination control, including examples described in the following sections.

13.4.5. Protecting Astronauts and Designing Human Rated Systems

According to Article IX of the Outer Space Treaty, 'appropriate measures' are required to avoid 'adverse changes in the environment of the Earth which could result from the introduction of extra-terrestrial matter.' Minimizing exposure of astronauts to potentially hazardous Mars materials, as well as monitoring astronaut health and microbial populations carried by human missions, are key factors in facilitating the safe return of astronauts to Earth. Comprehensive monitoring is essential, to ensure adequate documentation throughout the mission in order to provide confidence that in-flight illnesses or other potential biohazards are of Earth origin. This will be required for planetary protection purposes at a level significantly greater than that needed to document astronaut health alone.

Particular aspects of human missions needing attention both to avoid back contamination and ensure human safety include the choice of initial landing sites, design of human habitats, plans for EVA's and potential ISRU operations. In general:

- Landing sites shall be selected such that nominal or off-nominal mission operations have a low probability of allowing mission-associated microbial or organic contamination to enter Mars 'Special Regions' either horizontally or vertically. This includes mission-induced Special Regions.
- Human habitation modules (and associated life support and recycling systems) should be located and operated in ways to ensure that mission-associated microbial or organic contami-nation has a low probability of entering 'Special Regions'.
- Human EVA's should be planned to likewise minimize mission-associated microbial or organic contamination of Special Regions (e.g. via robotic access beyond designated zones of minimal biological risk (ZMBRs) as recommended by the NRC, 2002b), and
- ISRU activities should be planned to avoid contamination of 'Special Regions' while also protecting humans and human-associated systems from uncontrolled contact with martian ma-terials from those regions.

These guidelines translate to many specific planetary protection technology needs including the development of human habitat egress/ingress procedures that minimize contact with Martian material; medical monitoring procedures before, during and, after EVAs; traverse planning in relation to "Special Regions"; monitoring of the microbial inventory of human habitats; protocol development for laboratory facilities on Mars; and development of quarantine procedures for affected astronauts. Assuming that a future Mars mission adopts a scenario many weeks or months on the planet – and EVA's as often as every other day, - this could mean many dozens of ingress/egress operations and transfers between habitat and lab areas, each of which has PP implications [Hogan, 2006]. Other technological systems processes with potential planetary protection implications include those associated with advanced life support systems, recycling waste disposal and ISRU. While COSPAR policy guidelines for the human exploration of Mars provide the framework to support requirements in many of these areas, specific implementation approaches will most usefully be developed in the context of anticipated technology developments in human health monitoring and molecular environmental microbiology.

Furthermore, there are important areas of basic scientific knowledge related to planetary protection that have cross cutting impact for development of effective human rated systems. Areas relevant to planetary protection that should be incorporated into the research portfolio include a) basic research to develop and extend our fundamental understanding of human associated and environmental microbiology in space; b) applied research and development of spacecraft hardware and systems to facilitate end-to-end mission capability; and c) testbed studies to evaluate effective implementation of systems, processes, crew training, and operations that address planetary protection requirements, as described below.

1. *Fundamental Knowledge on Microbial Limits of Life, and Human-Associated Microbial Diversity and Distribution:*

In the past two decades, our understanding of environmental microbiology and extremophiles has expanded considerably, resulting in a greater awareness of the potential for the survival of terrestrial microbes in extreme environments, as well as the prospect for finding possible evidence of truly extraterrestrial life in other locations. Faced with such possibilities, it is essential to the proper implementation of planetary protection policy that criteria for assessing habitability for planetary environments are established conservatively, and that appropriate measures are taken to protect against contamination. Thus, it is essential that research on microbial diversity and adaptation to planetary environments continue to inform planetary protection policies and their implementation, for both robotic and human missions.

Furthermore, we have only recently recognized that humans themselves are a veritable scaffold upon which microbial ecosystems flourish. Powerful new analytical tools have become available to analyze and decipher such ecosystems and understand our human associated micro-organisms [e.g., Stone, 2009][33]. Since these diverse microbial hitchhikers represent unavoidable potential bio-contaminants during human exploration in the solar system, it is important to understand them to the fullest—their identities, abundance, and distribution, as well as their potential for dispersal, survival and propagation as contaminants, and as markers in exploration environments, whether in habitat/work environments or exposed to the planet/moon environment.

Specific topics of relevance to the fundamental scientific understanding of biological and physical sciences in space include (but are not limited to):

- Development of a baseline inventory and understanding of human associated microbes, as relevant to the space environment;
- Studies of human associated microbes as potential contaminants, including their abundance, potential for release, and dispersal/survival/propagation in human planetary exploration;
- Understanding human associated microbes as potential biomarkers of relevance, and their possible use as tracers of contamination;
- Survival of spacecraft relevant terrestrial organism and molecular components;
- Contamination transport models and pathways (near- and far-field);
- Studies to understand the contribution of ambient space environments towards passive mitigation of forward contaminant risks (radiation, temperature, etc.).

2. *Applied Research and System Development*

Parallel developments in applied fields are needed to provide capabilities encompassing the entire spacecraft hardware system, including processes and procedures and the human interface:

- Development of monitoring technologies to evaluate the level/type of microbes released by human associated activities on an ongoing basis, with capability for monitoring microbes in real-time, integrating system technologies to protect human life from pathogenic and /or alien microorganisms (should they exist), and shielding engineering systems from bio-corrosion;
- Development of human quarantine and decontamination strategies and capabilities for planetary environments, aiming to minimize exposures and control recontamination parameters;
- Quantitative and qualitative analysis and understanding of process streams of life support systems (air, water, recycling wastes, etc.) from a human microbiology perspective, for all crew rated systems, including an end-to-end understanding of venting, dispersal, and shutdown considerations to minimize release of contaminants and enable surface containment/disposal of wastes;
- Development of sterilization and decontamination capabilities for generated wastes, spacecraft volumes (habitats, labs, pressurized rovers etc.) and associated equipment and samples, consistent with available resources anticipated for such missions;
- Assessment and understanding of nominal contaminant releases from cabin atmospheres/other enclosures via leakage;
- Development of responses to off-nominal scenarios and contamination events, with implications and mitigation requirements for both planetary protection and crew health/safety objectives.

3. *Test beds for Technology Development and Operations*

The Moon in particular is considered to be an excellent potential testbed to develop planetary protection procedures and practices in an environment sufficiently harsh to prove an adequate challenge, but—unlike Earth analogues-- isolated from the overwhelming background contamination of the terrestrial biosphere. Because the Moon is currently recognized as being of interest for understanding pre-biotic chemistry and the origin of life, but is not hospitable to contamination by Earth life, the only current planetary protection constraint for operations on the

Moon is the requirement to document activity. On the Moon, there are no limits on contamination similar to those in place for more distant but potentially habitable bodies such as Mars. This means that technologies developed for use on the Moon are not prohibited from releasing high levels of contamination per se. A coordinated lunar program addressing planetary protection issues could yield significant benefits [e.g., LEAG, 2009][34] such as providing valuable ground truthing on in situ contamination of samples and external environments; studies of lunar habitat/spacesuit competency, containment and leakage; and testing operational procedures associated with successful planetary protection implementation on another planetary surface.

Science Investigations on Mars: On robotic missions to Mars that study the potential for extraterrestrial life, strict PP controls are imposed to avoid forward contamination of the planet by biological organisms from Earth. These requirements are fully integrated into MEPAG Goals I-III—which seek to provide necessary precursor information for future human missions [MEPAG, 2008][35]. Experience with past robotic missions has informed the development of requirements and implementation options that are explicitly detailed in COSPAR policy and NASA Procedural Requirements document NPR 8020.12 (currently, version D) [NASA, 2011][36]. For landed hardware, compliance with these requirements involves rigorous bio-burden reduction and accounting pre-launch, and operational constraints through end of mission. Hardware involved in the acquisition and storage of samples from Mars must be designed to protect Mars material from Earth contamination, and ensure appropriate cleanliness from before launch through return to Earth. Technologies needed to ensure sample cleanliness at levels similar to those maintained by the Viking project, are yet to be developed in the context of modern spacecraft materials and human missions. Maintaining appropriate separation of habitat and laboratory modules will also be important design elements for science investigations. Obviously, attention to these planetary protection considerations for science objectives will remain important aspects of future human missions.

The objectives of planetary protection policy are the same for both human and robotic missions; however, the specific implementation requirements will necessarily be different. MEPAG Goal IV + recognizes that additional forward contamination R & TD will be needed for operations and collection of samples during sustained human crews presence. Human missions will require additional planetary protection approaches that minimize contamination to martian environments released due to human exploration, including protocols on how to access locations on Mars (both characterized and un-characterized) and performance standards for human support systems, including lab handling and testing of pristine materials on Mars. Robotic elements of human missions must still follow relevant planetary protection requirements: for example, access to Mars “Special Regions” (as defined in NPR 8020.12D) involves stringent cleanliness requirements, which will also necessitate targeted technology development for both hardware cleaning, reuse, and clean transfer capabilities. A conceptual approach is presented in NASA's Draft Reference Architecture 5, but implementation requirements for human-based exploration must be refined in the context of specific planned missions, especially considering the extensive drilling anticipated to many tens of meters below the surface. In addition, preparation and placement of pre-landed assets and hardware (including nuclear power systems and fission reactors that must avoid creation of mission induced special regions from radiated heat) will likely require special considerations, both for forward contamination concerns as well as ‘Special Region” avoidance. Finally, advanced technological and operational considerations will be required on how to respond to a discovery of putative martian life, if found or detected during a human mission.

Protecting Earth: Preventing adverse effects on the Earth's environment as a result of returning astronauts and/or samples from Mars is the highest planetary protection priority in COSPAR policy and guidelines, in accordance with Article IX of the Outer Space Treaty. Requirements for Mars Sample Return (whether via robotic or human missions) involve stringent restrictions on release of unsterilized Mars material into the Earth environment. The European Science Foundation (ESF) has recently completed a study on assuring the safety of robotic Mars sample return missions [ESF, 2012][37] and has endorsed previous guidance that the constraints be formulated as an assurance level for the release of a particle of martian material of a size that could potentially carry biological hazards. Specific numerical requirements recommended by the ESF involve ensuring that a particle of unsterilized Mars material is contained with a probability of 1x10-6. In consideration of new information about viruses and genetic transfer agents, the particle size limit recommended by the ESF study for containment is 10nm. Containment at this level is required until samples are characterized and demonstrated to be safe for release, which includes satisfactory completion of a life detection protocol, although re-allocation of the full 'probability of release' is anticipated upon successful introduction of the return capsule into an Earth-based containment facility. Similarly, the direct return to Earth of the MTV and crew has planetary protection implications, just as during the Apollo missions. Thus, technologies for life detection, biohazard avoidance and the protocol for use during human missions are also areas in need of further development and technical refinement.

The Apollo Program provides a cautionary example of how NASA has implemented planetary protection on human missions in the past, for both astronaut and samples—and illustrates numerous targets for making improvements to future mission designs and procedures. Among the key technologies for human missions to Mars include the selection of appropriate spacecraft materials and hardware; the design of suitable human rated subsystems; the development of procedures for clean sample acquisition, handling, and containment; capabilities to ensure adequate re-cleaning of sampling hardware in situ; and quarantine capabilities for astronauts both on Earth and during return from Mars. While detailed protocols have been developed for testing and handling pristine martian samples returned via robotic missions [Rummel, 2002][38], the eventual protocols for returning samples via human missions have yet to be developed.

13.4.6. Planetary Protection Conclusions:

In considering future human exploration to varied destinations beyond Low Earth Orbit (LEO), concerns about harmful contamination and planetary protection (PP) will have significant impacts on mission architecture, requirements, capabilities and activities. Already, for human missions to Mars (and the Moon as test-bed), planetary protection concerns are recognized as introducing cross-cutting technological and design challenges for spacecraft and vehicle systems; habitats and labs; EVAs and suits; science exploration and operations; equipment cleaning, maintenance and use; human and robotic access to 'Special Regions'; quarantine/containment protocols; and even shutdown/dormancy of infrastructure between missions. Obviously, these are not just add-on or 'other' issues for later consideration; rather they're integral to planning and design from the start.

While precise PP protocols will be developed in coming years for human missions to Mars and other target bodies, it will be important that mission architects, designers and engineers become aware of international PP policies, which are not optional and are subject to revision in the face of new scientific information. Application of planetary protection policy in the context of human missions beyond LEO is necessary for adherence to the Outer Space Treaty, just as it is for robotic missions. Attention to PP in the early architecture design and planning stages will help use common solutions where possible, thus avoiding unnecessary duplication of efforts and costly redesign of critical elements and systems.

Through organized workshops and interdisciplinary information exchanges, the planetary protection community has begun to explore with engineering and systems experts the impacts of COSPAR and NASA planetary protection policies on numerous human associated systems. By establishing communication among different groups, previous planetary protection studies and workshops can be useful in highlighting important data needs, as well as identifying priority R&TD areas. As systems experts develop the next generation of plans, design elements and operation scenarios, they will benefit greatly from consulting this information from the start. Finally, compliance with PP policy is important for overall mission success and public support. NASA's commitment to transparency and participation for future human missions must certainly include disclosure of risks as part of mandated environmental impact reports as well as all public engagement activities. Planetary protection is thus an essential element in any architecture for future human Mars exploration and must be incorporated from the start.

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